X-734-67-475

NASA TM X-63018

RADIOISOJET PROGRAM SUMMARY REPORT

SEPTEMBER 1967

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

*This Report summarizes a joint program between the Goddard Space Flight Center and the United States Atomic Energy Commission.

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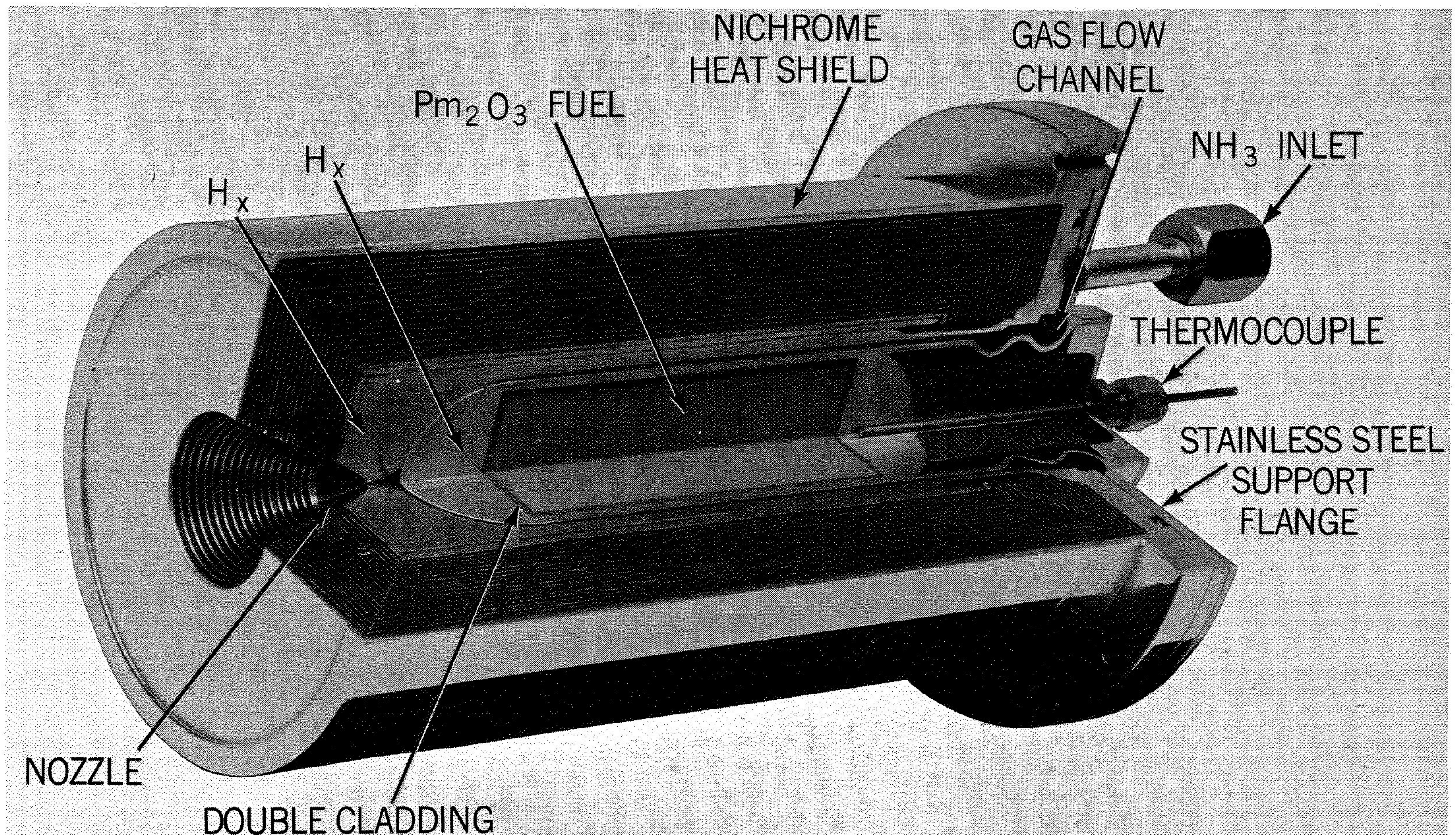

Frontispiece. The **R**adio**I**so**J**et (RIJ)

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I. ABSTRACT

The primary goal of this ~~summary~~ report is to present the results of the Radioisojet (RIJ) program, which successfully demonstrated the feasibility of the RIJ concept. ~~The microthruster, although conceived at NASA-Lewis in the~~ form of the resistance jet, evolved to its present configuration ~~utilizing a radio~~-isotope fueled heater at the NASA-Goddard Space Flight Center.

The RIJ design approach was to simulate the thermal storage resistance jet concept, wherein the thruster body temperature is maintained by a heat shield package. Thrust is obtained by passing a propellant (NH_3) over the isotope heated capsule surface in short pulses. A thrust advantage is gained by causing the propellant to decompose as it traverses the fuel capsule surface. The decomposition process is enhanced by constructing the fuel capsule with a suitable catalytic material. Hastelloy-X was used as the outer capsule surface material for this purpose. Additionally, the thruster design incorporated an interchangeability feature which allows the radioisotope fuel capsule to be replaced by an electrically heated capsule. In this manner base-line and system shake-down tests may be performed using the electrically heated RIJ.

Initial thruster tests were performed at both the General Electric Company's vacuum test facility, Evendale, Ohio and the Mound Laboratory (Monsanto Research Corporation), Miamisburg, Ohio test facility utilizing an electrically heated thruster. Subsequent to the electrically heated tests a hot-test firing sequence was conducted at Mound Laboratory utilizing the radioisotope heated thruster. The hot-firing test sequence was performed over approximately a 30 day period. No radiological or other safety incidents occurred during the test program.

The heat source for the RIJ comprised approximately 60 thermal watts of highly purified and aged promethium-147. The capsule development and fabrication work was performed by the Battelle Memorial Institute, Pacific Northwest Laboratory, Richland, Washington.

The program can be summarized as successful. Most program objectives were achieved and the program was conducted in a timely manner. A performance capability of 232 seconds of specific impulse at a nominal gas temperature of 1500° F and at a thrust level of 20 milli-pounds was demonstrated. Post test evaluation has resulted in the definition of additional work tasks and tests which are recommended for future study to attain the ultimate goal of flight developed RIJ hardware.

II. INTRODUCTION

Reactive force requirements for spacecraft attitude control and station keeping for synchronous orbits have introduced a new dimension in thrust system technology. These requirements are characterized by the very low levels of thrust required, the intermittent demand for thrust pulses, and the long lifetimes demanded. Conventional chemical systems have inherent limitations for these applications; ion engines still demand electrical energies inconsistent with spacecraft power supplies. Cold gas systems have many advantages from reliability and simplicity points of view, but their low efficiencies necessitate quantities of gas inconsistent with long mission lifetimes. This situation led the National Aeronautics and Space Administration-Goddard Space Flight Center to embark on a program to determine the feasibility of using hot gas systems for these applications, the relative merits of various heating techniques, and their practicality for these applications.

A. BACKGROUND

Work on the resistance heated hydrogen jet was begun at the NASA-Lewis Research Center in 1959 (Reference 1). Feasibility was demonstrated and a patent disclosure was made on April 4, 1960. The initial engine was designed for a thrust of one pound with a target specific impulse (I_{sp}) of 1000 seconds, high overall efficiency and long operating life. To achieve this performance, a heater temperature over 5000°R and a power input of approximately 30 kilowatts was necessary for continuous jet operation. The approach employed at NASA-Lewis was to put this engine on the intermediate (observatory) class of spacecraft; however, the heater temperature and power requirements posed too severe a problem. For example, very few materials are available today as heater elements that can withstand an operating temperature of 5000°R, especially for long periods. In addition, the problems faced in getting aboard a spacecraft were compounded because of the weight and size requirements associated with storing liquid hydrogen, and carrying power equipment to supply and control 30 Kw. Further, the design for the continuous operation mode was not optimum for attitude control and station keeping since these functions can best be performed by systems working in the pulse mode.

In 1962, W. Isley of the Goddard Space Flight Center (GSFC) proposed to minimize the electrical power requirement by using a radioisotope heater, and made analytical studies to determine the potential performance of such a system using liquid hydrogen, liquid helium or ammonia as the propellant gas. The analysis predicted performance with radiation-cooled thrusters (at realistic

temperatures) ranging from an I_{sp} of 300 seconds for ammonia to 700 seconds for liquid hydrogen.

A GSFC in-house task, initiated in 1964 to cover initial studies, was directed to the definition of conceptual designs for three requirements: attitude control, station keeping, and orbital transfer associated with attainment and operation in a synchronous orbit. Primary to this effort was the collection and compilation of all available radioisotope fuel data. These data were used in thermal analyses to determine mission life, materials compatability and potential systems performance. It was obvious that for some space missions a radioisotope could not be tolerated aboard a spacecraft due to radiation interference with onboard sensors. It was apparent also that certain radiation hazards would be attendant to ground handling (pre-flight, storage, shipping, etc.) of the radioisotope thruster system. However, it was established that thruster propellant consumption and thermal performance could be simulated with an electrically heated (resistance) jet. Thus, contract (NAS5-9013) was negotiated with the General Electric Company to produce resistance jets at station keeping (50 millipound) and attitude control (15 millipound) thrust levels.

The efforts in 1965 were directed toward the development of thermal models which, in conjunction with data from the resistance jet testing, were used to more accurately evaluate the problems and potential of the radioisotope heated thruster. On the basis of these studies and discussions with the United States Atomic Energy Commission (USAEC) technical personnel, it was concluded that radioisotopes with reasonably long half-lives, high specific power, and low nuclear radiation levels hold considerable promise as heat sources for microthrusters. A joint effort pilot development program to assess this concept was proposed by NASA Goddard Space Flight Center to NASA Headquarters and to the USAEC. With favorable reactions from NASA Headquarters and from the USAEC, steps were taken to implement the program. Following is a chronology of events pertinent to the current Radioisojet (RIJ) program.

PERTINENT CHRONOLOGICAL RIJ PROGRAM EVENTS

Event No.	Date	Activity (Location)
1	10/27/65	First Program Coordination Meeting (GSFC)
2	11/24/65	First Design Review Meeting (BNW)
3	1/24/66	Second Design Review Meeting (Mound Lab)

4	5/25/66	Third Design Review Meeting (GSFC)
5	6/10/66	GSFC/USAEC Design Approval
6	11/ 1/66	Pm-147 Fueled Thruster Delivered to Mound, Initiation of Fueled Thruster Testing
7	12/ 3/66	Completion of Fueled Thruster Tests
8	2/ 3/67	Final Program Coordination Meeting (GE)

B. PURPOSE OF THE TEST PROGRAM

The pilot program involved the development and testing of a simple single-pass hot gas microthruster which employed a radioisotope capsule. The overall objectives of the program were the demonstration of the feasibility of employing a radioisotope heater, a demonstration of performance characteristics, and the identification and solution of thruster and capsule material and construction problems.

The initial thruster performance specification (in the pulse mode) was defined as follows:

(1) Thrust: 0.020 pounds-force (attitude control)

(2) Operating Temperature: 2000° F Max. (60-70 thermal watts)

(3) Propellant: Ammonia (NH_3)

(4) Radioisotope: Promethium-147 Oxide (Pm_2O_3)

(5) Fuel Capsule L/D: 3 (Max.)

The design philosophy was based on the development of thruster bodies and capsules separately. The thruster body design would utilize previous experience with electro-thermal (resistance) jets, and would include a removable electrical heater to provide thermal, propulsive performance, and power consumption data prior to integration of the thruster body and radioisotope capsule.

Although "hot testing" of propulsive devices and systems is an accepted technique for developing and qualifying flight hardware, such was not the purpose for this program. Despite the evidence of thermodynamic analysis and simulated testing using electro-thermal (resistive) heaters, there were so many imponderables that a specific verification of principle and demonstration of concept feasibility was deemed absolutely necessary before embarking on an expensive program of flight hardware development.

Specific objectives of the RIJ Program can be summarized as follows:

(1) Demonstrate concept feasibility,

(2) Establish performance capabilities,

(3) Define the facilities and procedures for testing a radioisotope fueled thruster,

(4) Define the radiation field and ground handling problems associated with a radioisotope fueled thruster,

(5) Define the measurements and data needed to establish thruster performance,

(6) Establish the appropriate division of responsibility between agencies and contractors (i.e., mechanical, electrical, managerial, interface).

C. PARTICIPANTS AND FUNCTIONS

The joint RIJ program between the Goddard Space Flight Center and Atomic Energy Commission was coordinated by NASA/GSFC through the NASA/USAEC Space Nuclear Propulsion Office and has been implemented with the assistance of NASA-GSFC contractors:

(1) General Electric Company, Missile and Space Division, Evendale, Ohio (as part of Contract NAS5-9670),

(2) Hittman Associates, Inc., Baltimore, Maryland (as part of Contract NAS5-10235);

and USAEC contractors:

(1) Battelle Memorial Institute, Pacific Northwest Laboratory, Richland, Washington (as part of Contract W-7405-ENG-92),

(2) Monsanto Research Corporation, Mound Laboratory, Miamisburg, Ohio (as part of Contract AT-33-1-GEN-53).

Program Management was effected by means of Coordination Conferences with representation from contributing governmental agencies and their contractors.

According to the interagency informal agreement, the USAEC and its contractors provided:

(1) The required radioisotope source, suitable encapsulated;

(2) Integration of a fuel capsule with a NASA developed thruster;

(3) Suitable facilities, including instrumentation for performing fueled thruster performance tests.

The USAEC was responsible for the maintenance of radiological safety throughout the program and for the analysis of results in all areas directly relating to radioisotope handling and capsule design.

The Goddard Space Flight Center and its prime contractor, the General Electric Company, were responsible for:

(1) The definition of the overall thruster specification,

(2) The design and construction of required thruster bodies to accept the fuel capsule,

(3) The provision of an equivalent electrically heated thruster for use in facility checkout,

(4) The provision of a thrust measuring system,

(5) The definition of thruster operational test requirements, and

(6) Analysis of test data to determine performance.

The General Electric Company also assumed responsibility for assuring the adequacy of instrumentation to meet operational test requirements, for calibration and checkout of instrumentation, and definition of the operational test program.

Hittman Associates, Inc., by arrangement, coordinated and edited this summary report according to the desires of the GSFC. All technical material and test data were provided by the major contractors, i.e., GE, Battelle, and Mound Laboratory. GSFC provided the Introduction.

III. THRUSTER — CAPSULE DESIGN AND INTEGRATION

A. THRUSTER DESIGN

1. Design Concept

The TSK 2000-1RE radioisojet thruster design is based upon the thermal storage resistance jet concept wherein the inner thruster body is maintained at operating temperature with a minimum power input by surrounding the system with an effective heat shield. During a pulse of propellant, the temperature level of the gas is increased by its contact with the hot surface of the flow path. With short pulses (milliseconds to a few seconds), the associated decrease in thruster body temperature is small. With longer pulses, the temperature drop of the thruster body is correspondingly larger since the thruster design incorporates a fixed or constant power heat source.

The thruster design, shown schematically in the Frontispiece and in detail in the engineering drawing of Appendix A-1, consists of three basic components: heat source and inner thruster body, outer thruster body and nozzle, and heat shield assembly. These components are shown in various stages of assembly in Figures III-1 through III-4. With this design, the outer thruster body and heater (inner thruster body) can be bolted together and this subassembly, in turn, can be inserted into the heat shield package; or the outer thruster body and heat shield can be bolted together and the heater inserted into the thruster-heat shield assembly. With this versatility, component replacement becomes a simple procedure.

The propellant flow path is formed by inserting the inner thruster body and heater with its attached flange and convoluted surface into the outer thruster body, nozzle, and flange subassembly. The two flanges are bolted together with an Apex seal between to maintain a leak tight assembly. The convoluted section of the heater support serves two purposes: first, to allow for perpendicular misalignment of the inner and outer thruster bodies to their support flanges, and secondly, to eliminate stresses caused by differential thermal expansion of the two coaxial cylindrical sections.

The propellant flow path is through the 0.25 inch diameter propellant feed tube, through a slot machined into the heater support flange, around the convoluted section and over the heater body surface through a 1.2 inch diameter 0.020 inch wide annulus. The propellant flow continues over the domed section of the heater, which is kept from intimate contact with the outer thruster body by three ribs machined into the nozzle end piece (shown in the photo of

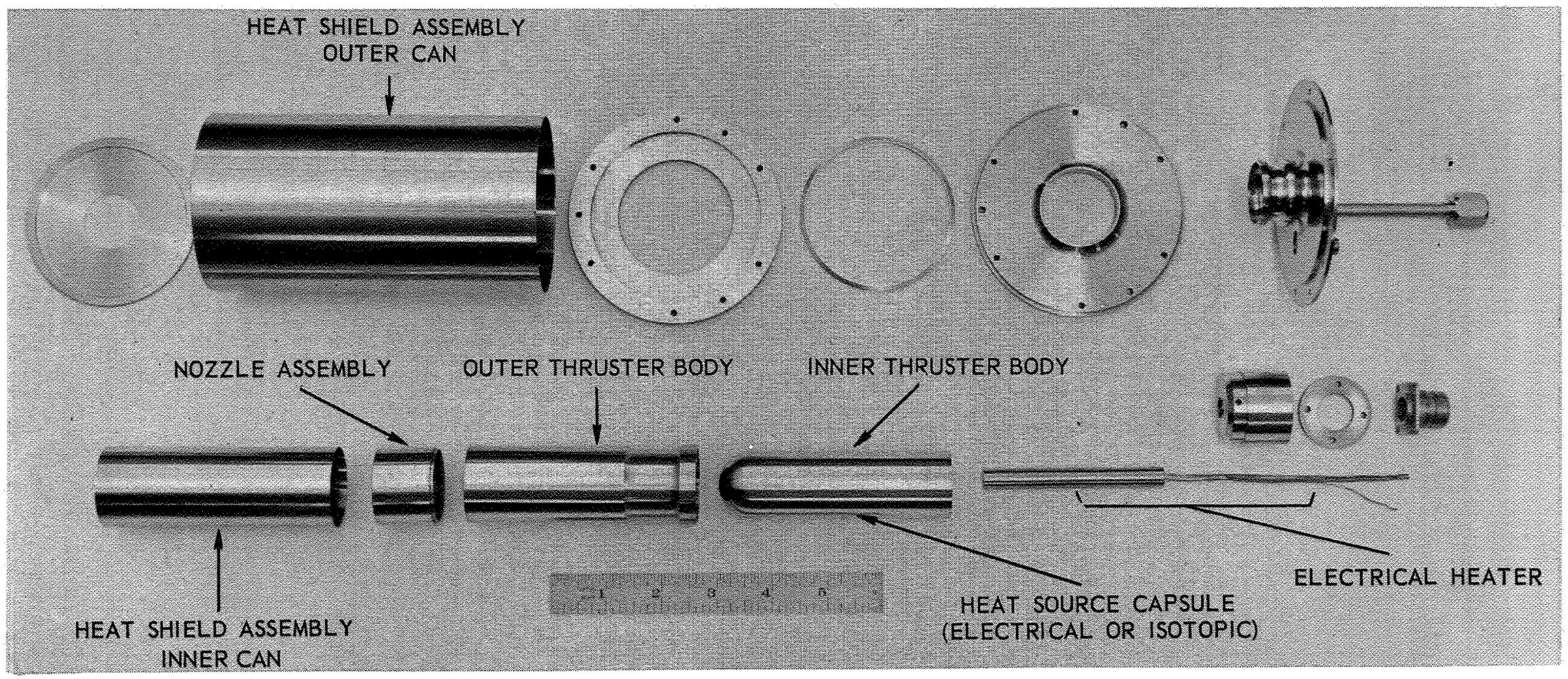


Figure III–1. Disassembled Thruster Components

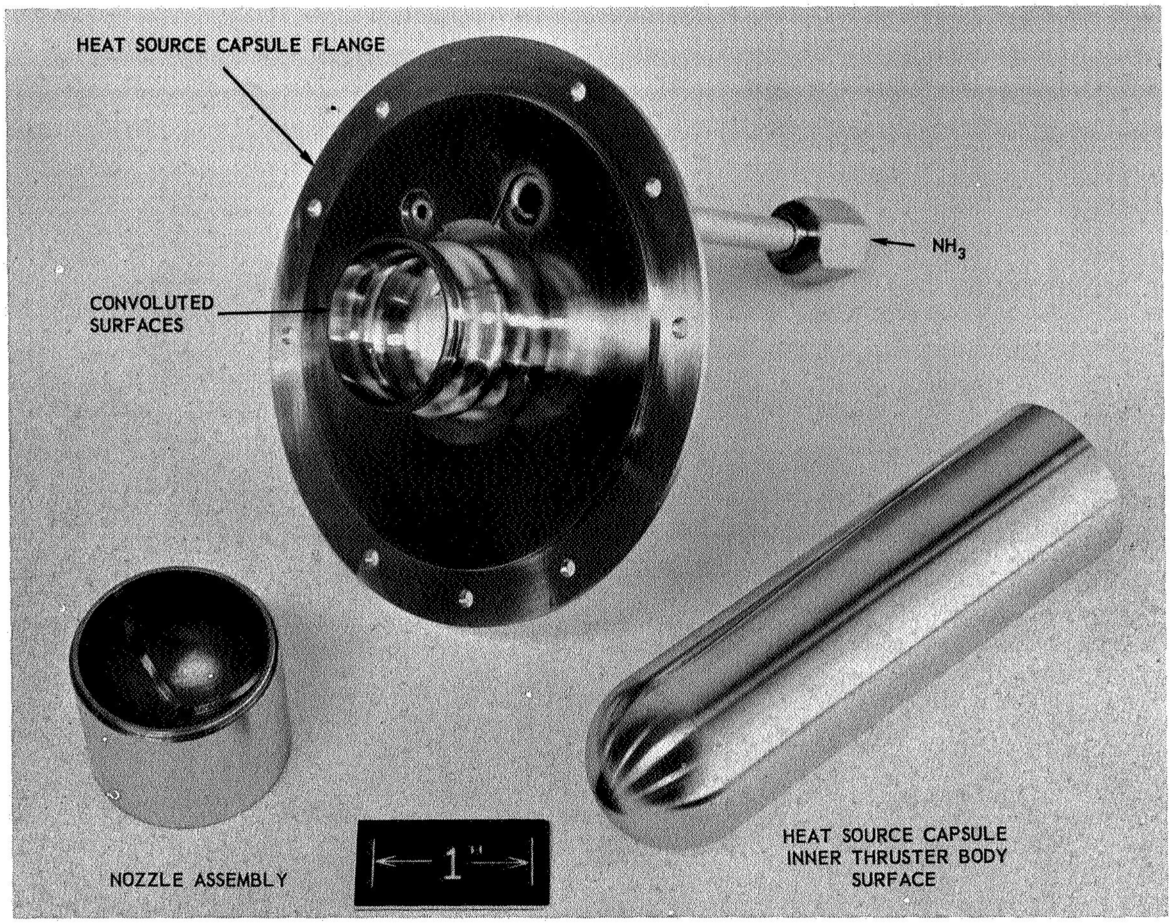


Figure III-2. Thruster Components

Figure III-2), thereby allowing room for propellant to flow around to the nozzle inlet section.

The exhaust nozzle is designed to provide 0.020 pounds thrust at an inlet pressure of 1.5 atmospheres. While the physical dimensions of this nozzle give an area ratio of 64/1, the actual area ratio, reduced by buildup of the boundary layer, is calculated to be no greater than 38/1. The configuration, optimized by utilizing a G. E. developed computer program, provides the maximum specific impulse obtainable within the constraints of the nozzle throat diameter and length.

Early in the design phase of the program, low emissivity materials, such as molybdenum, and low emissivity coatings, such as gold, were considered for use in the heat shield package. After some preliminary tests with molybdenum

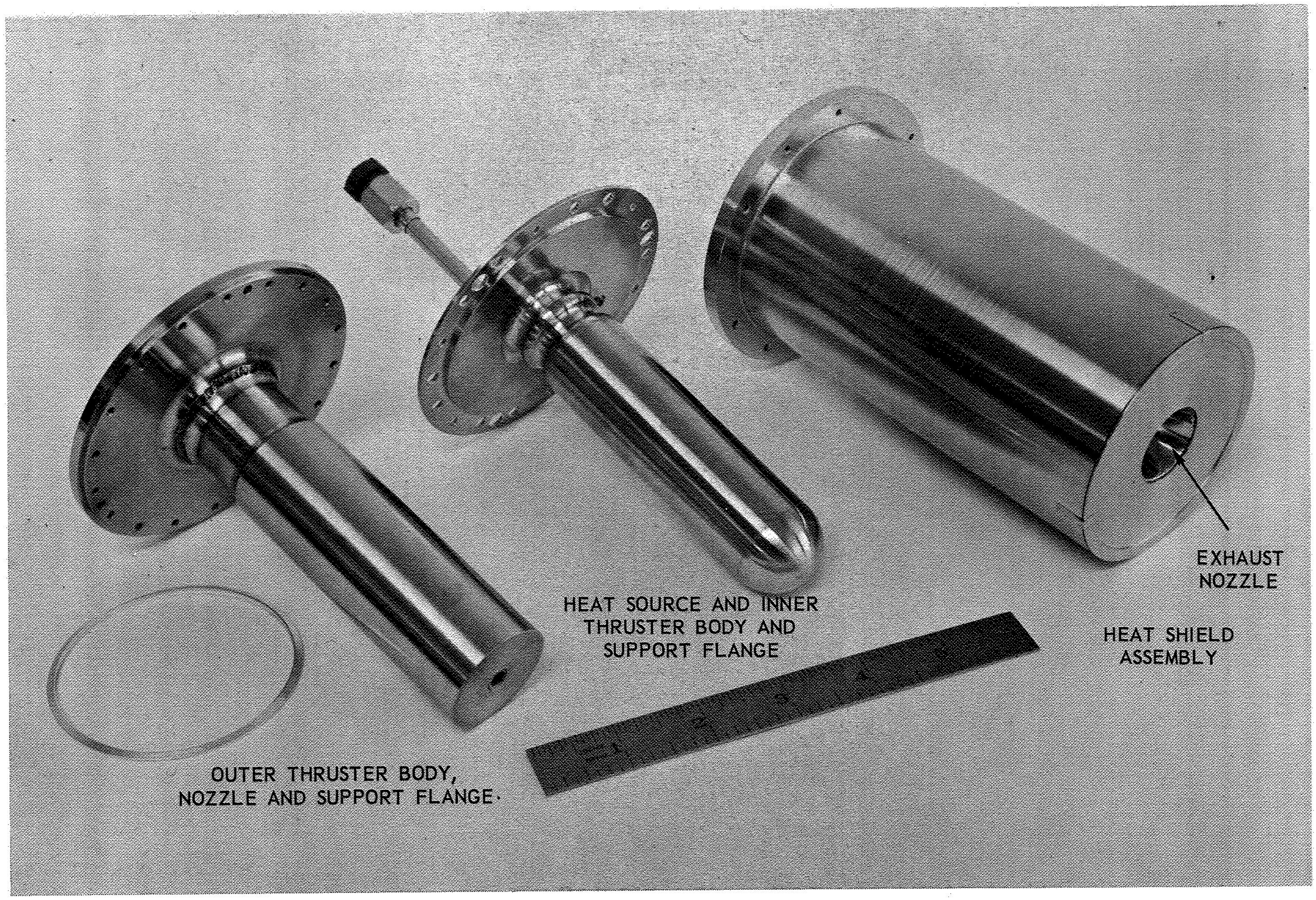


Figure III–3. Major Thruster Components

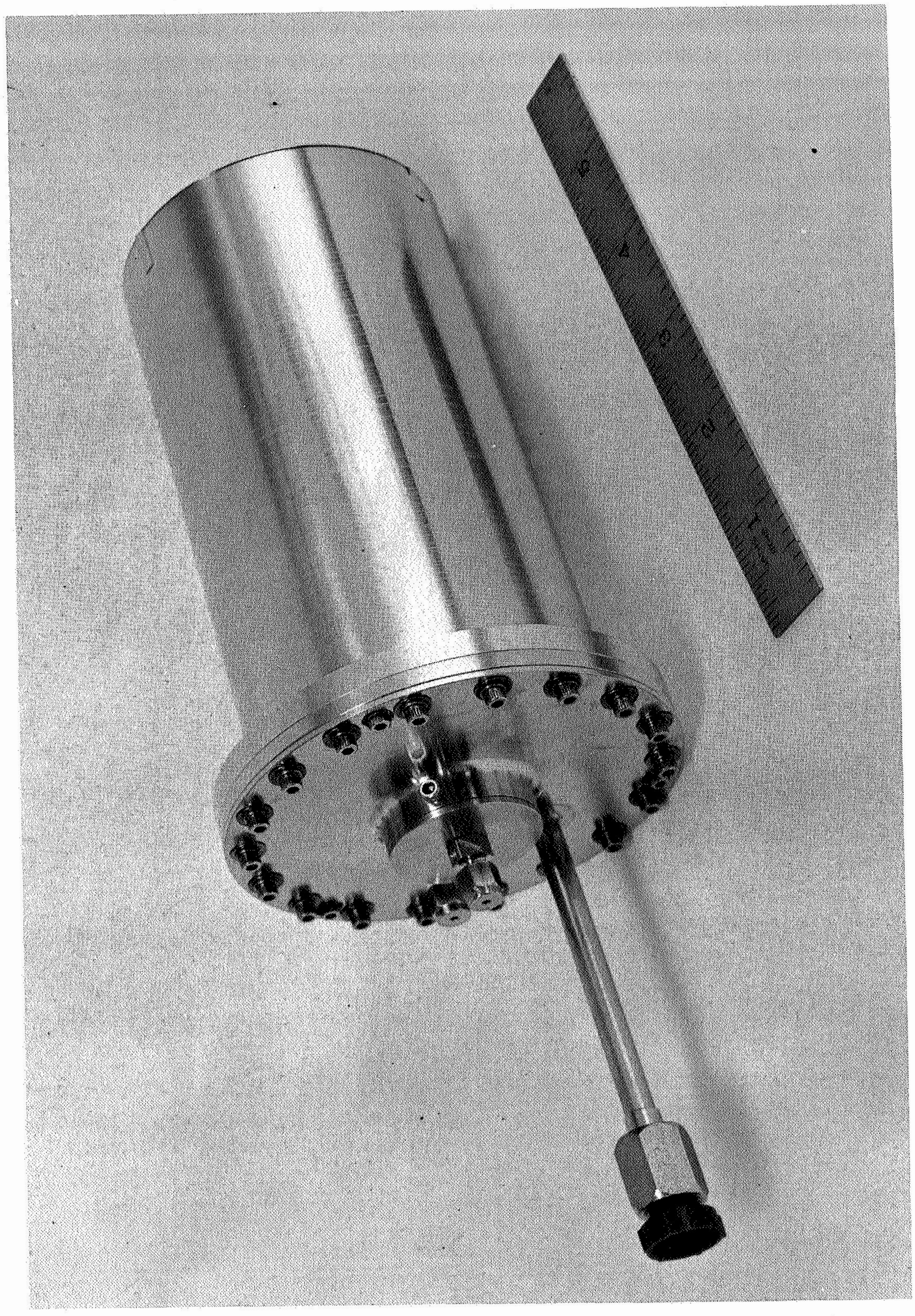


Figure III-4. Assembled Microthruster

foil, it was concluded that its lack of surface reproducibility and inherent difficulty of fabrication created problems which were not solvable within the scope of this program. For this reason, the Nichrome V foil and wire separator was chosen for use in the final design. The anticipated thermal loss from the system operating at 2000°F core temperature was calculated at 55.5 watts, based on estimated surface temperatures. The actual thermal loss based on measured temperatures and a core temperature of 1900°F is calculated to be 66 watts. This reflects close agreement with the observed electrical power of 65 watts needed to maintain a core temperature of 1900°F.

A study was performed to optimize the L/D of the heater capsule on a minimized heat loss basis. The results of this study indicated little difference in heat loss between an L/D of 2/1 and an L/D of 3/1. The outer diameter of the heat source capsule was fixed at 1 inch, which, with a required volume of 2 cubic inches, resulted in a 2.56 inch length or an L/D of slightly over 2.5/1. The remainder of the thruster configuration was built around this volume. The maximum diameter of the thruster assembly at the mounting flange is 3.95 inches, and the maximum diameter of the heat shield package is 3.13 inches, and its length, exclusive of connectors and propellant feed tube, is 6.6 inches.

The completed thruster (Figure III-4) with an electrically simulated heater weighed 5 lbs. While no direct measurement of the radioisotope fueled thruster weight was made, it is calculated to be only slightly heavier (approximately 5.4 pounds) than the electrical thruster.

2. Materials

The materials of construction of the TSK 2000-1RE thruster are essentially the same as those used in typical resistance jet thrusters developed for GSFC under Contracts NAS5-9013 and NAS5-9670. Thruster body material has been Hastelloy X primarily because of its strength at 2000°F and its ability to function effectively as an ammonia decomposition catalyst at that and slightly lower temperatures. A literature search performed during early resistance jet development work had indicated that, of the non-refractory and non-precious metal groups, the combination of molybdenum and iron found in Hastelloy X provided the best combination for effective catalysis at 2000°F (Reference 2).

With the requirement established to maintain a gas side surface of Hastelloy X in the hot zone, it was necessary to fabricate the inner thruster body surrounding the capsule of Hastelloy X also. The flanges and support tubing (being in a colder area and not contributing directly to decomposition) were fabricated from 304 stainless steel which welds readily to Hastelloy X. The heat shield layers were fabricated from 0.003 inch thick Nichrome V sheet separated by

0.014 inch diameter Nichrome V wire. The assembled heat shield package was inserted into a 0.020 inch thick 304 stainless steel cylinder brazed to a mounting flange of the same material and an end plate welded in place to support the assembly.

The seal used between the thruster body flange and the heater flange is a "V" cross section circular seal of silver plated 18-8 stainless steel. With this cross section, internal pressure tends to expand the "V" and force the relatively soft silver against the stainless steel groove, deforming the silver and providing a leak tight seal.

The propellant feed tube and pressure tap fitting were Tungsten Inert Gas (TIG) welded to the heater support flange and back brazed with an 1850°F gold-nickel alloy for increased support.

B. THRUSTER HEAT SOURCE

The microthruster heat source capsule (as shown in Figure III-3) may consist of either the electrically simulated or the promethium-147 fueled capsule. Both configurations are identical along the Hastelloy-X envelope (inner thruster surface), but differ, obviously, within the actual heat source. The fueled heat source double capsule was designed for a slip fit into the domed Hastelloy-X cylinder (inner thruster body), shown in Figure III-5. The electrical simulator was manufactured from solid Hastelloy-X bar stock domed at one end and bored at the rear to receive a 3/8 inch diameter electrical heater. These electrical heaters, referred to as HSK 2000-70 heaters, are shown in Figure III-1 and in the radiograph of Figure III-6.

1. The Promethium-147 Fueled Capsule

The domed Hastelloy-X sheath seen in Figure III-5 contains a double clad promethium fuel capsule. The constraints on the design of the capsule heat source imposed by thruster operating conditions and interchangeability included:

Fuel Form: Pm_2O_3 with demonstrated cladding compatibility.

Fuel Loading: nominal 60 watts, thermal.

Radiation: minimum consistent with the design.

Heat Source Dimensions (max): diameter — 1.145 in., length — 2.745 in. The inner and outer clad materials were fabricated from a 99.99 percent pure

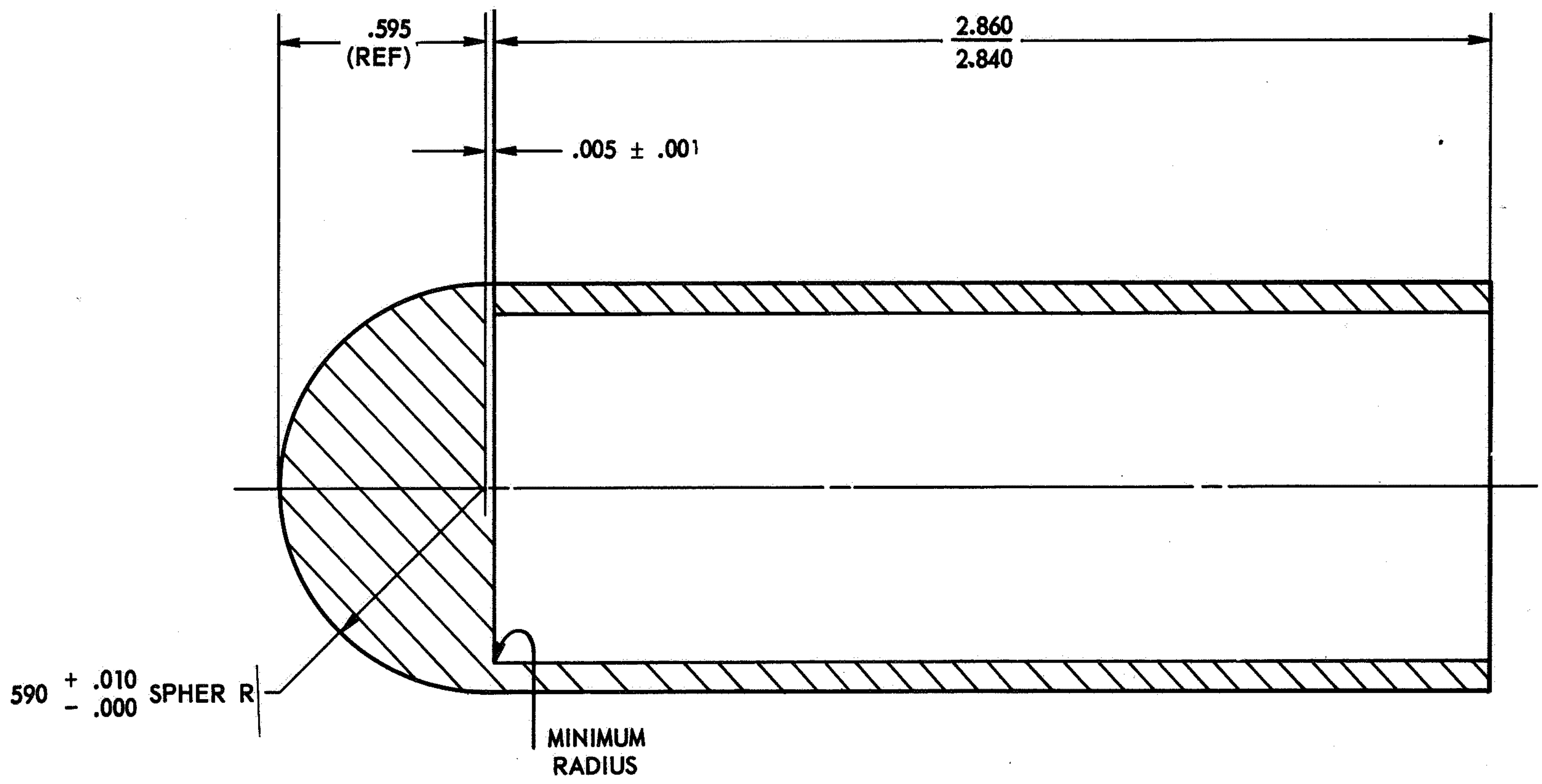


Figure III–5. Hastelloy-X Inner Thruster Body Cylinder

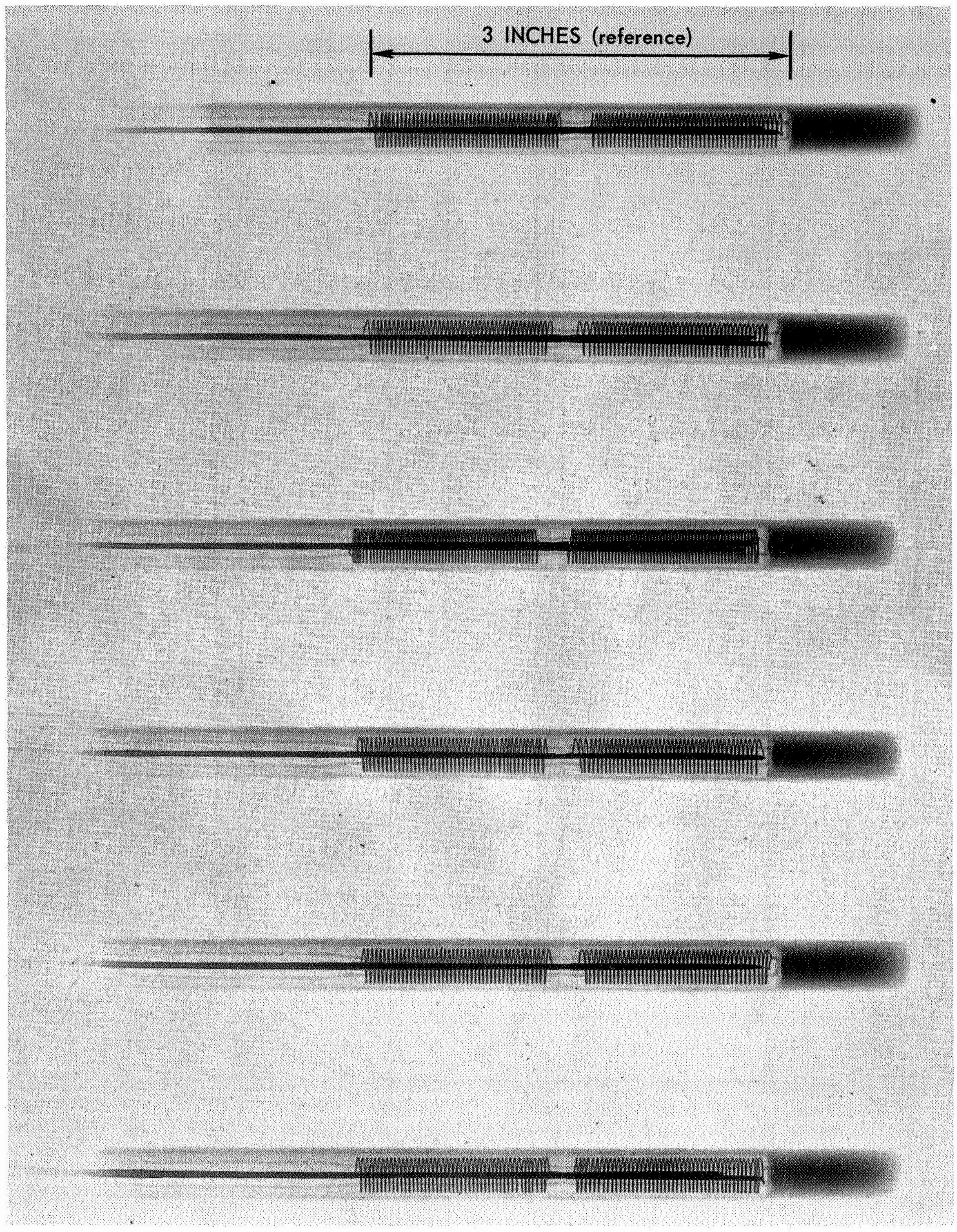


Figure III–6. Radiograph of HSK 2000-70 Electrical Heaters

powder metallurgy product which was subsequently rolled and seam-welded to form the capsules. Fuel-clad and metal-metal compatibility studies are discussed in greater detail in the classified appendix (D).

The seam welds on the cans were made by a semi-automatic tungsten inert gas (TIG) process. Peripheral TIG welds were made on the capsule (inner and outer clad) end closures (cf. Figure A-4, Appendix A). Electron beam (EB) welding techniques were also used in the fabrication and sealing of capsule components. The EB welder was used to drill outgas holes in the final endcap prior to assembly and to close these outgas holes (in vacuum*) after the TIG peripheral weld.

The inner and outer liners were originally supplied with bottom end caps of 0.060 inch thickness welded in place. These were replaced with end caps of 0.030 inch thickness to reduce the overall length of the finished capsule. Wall thicknesses for the final clads were 0.031 inch for the inner liner and 0.022 inch for the outer liner.

Extensive data were taken during the actual encapsulation of the fuel (Figure III-7). Data collected included radioactive contamination smears, helium leak, dye penetrant, ultrasonic, gamma spectrometry, calorimetry and dimension measurement tests on the capsules, and from dye penetrant, radiography and metallurgy of test welds. A description of each test is summarized in Table III-1. The results are summarized in Table III-2. The final doubly clad fueled capsule was 1.135 inches in diameter and 2.645 inches long.

2. Promethium-147 Fuel

Promethium-147 (half-life 2.62 years) is one of the rare earth uranium-235 fission products which may be obtained by chemical separation of spent reactor fuel utilizing a series of successive precipitation processes (References 3 and 4). Two major contaminants of promethium-147 are found: promethium-146 (half-life 5.5 years, Reference 5), and promethium-148 (which has two isomers with half-lives of 5.4 days and 42 days). After lag storage to allow for decay of the gamma emitting Pm-148, the rare earth crude is further refined to isolate the Pm-147 from its adjacent rare earth neighbors and all other impurities. A detailed discussion of this purification process and the subsequent oxide conversion process is covered in Appendix A-3.

*The vacuum within the capsule permits use of HERF (High Energy Rate Forming) which is a candidate final process to yield a high integrity, intimately bonded, multi-clad heat source capsule for space systems applications.

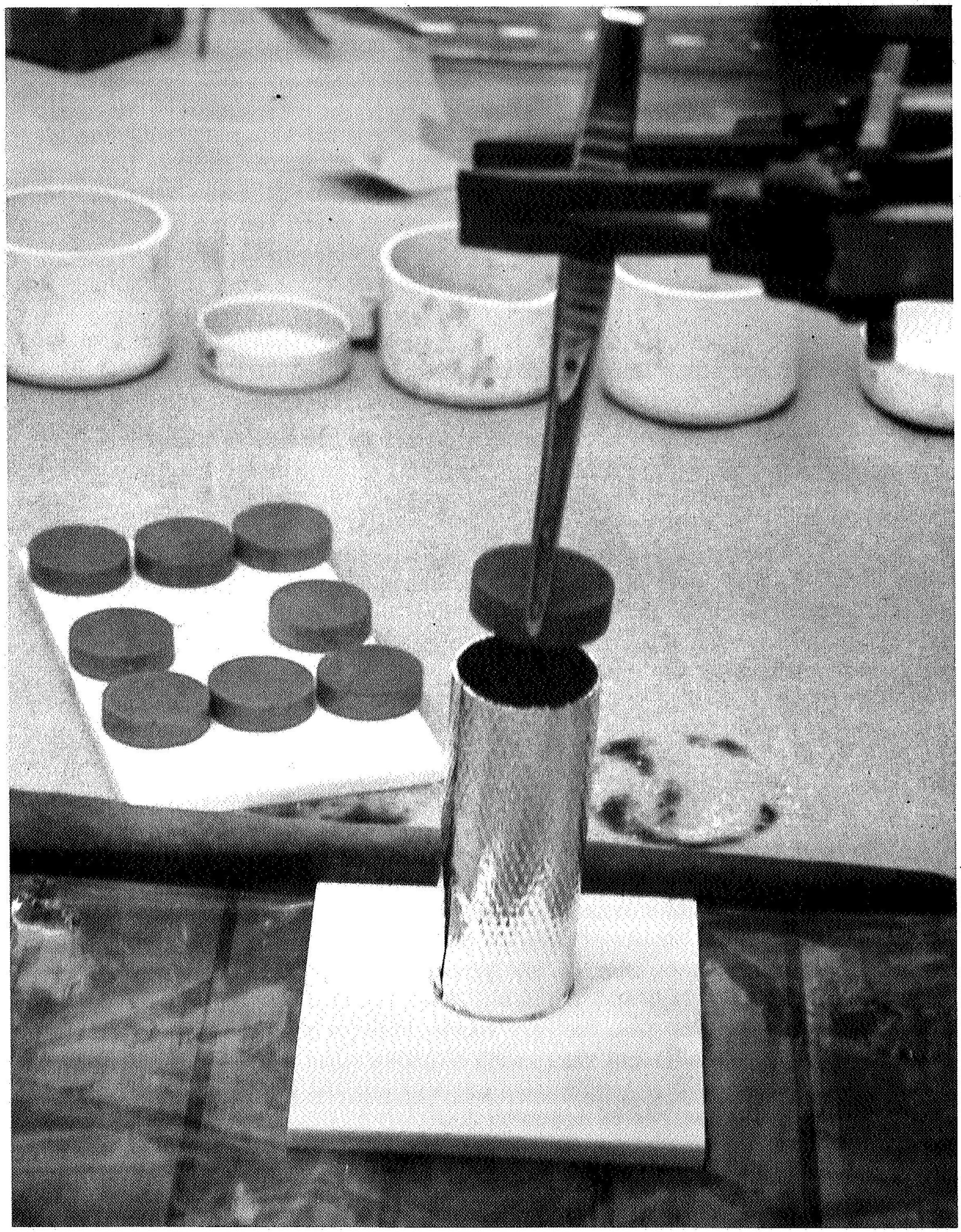


Figure III-7. Loading Pm_2O_3 Pellets Into Inner Capsule Liner

Table III-1

Promethium-147 Capsule Test Descriptions

1. Radioactive Contamination Smear Test:	Wipe capsule with a dry cotton swab and measure the contamination level with a mica window (Beta detection) GM counter. Decontamination is effected by wiping with moistened cotton swabs.
2. Helium Leak Test:	CEC helium leak detector, chamber containing the capsule evacuated to .05 torr and held for 30 minutes, backfilled with helium to 15 psi and held for 5 minutes; leak detection sensitivity checked before and after capsule leak check. Minimum detectable leak = less than 7×10^{-7} std. cc/sec helium.
3. Dye Penetrant Test:	Zy Glo Post Emulsifier technique (Magnaflux Corp.); viewed under ultraviolet light.
4. Ultrasonic Integrity Test:	Hanford developed broad band ultrasonic tester, 15 Mc Li_2SO_4 transducer 3/4" spherically focused with a mechanical system calibrated to ±.001 inch.
5. Radiography:	GE OX250 x-ray machine; Kodak type M film; viewed under high intensity illumination.
6. Metallography:	Four sections per weld, polished and etched.
7. Radiation Mapping:	CP dose rate meter calibrated with a known source.
8. Gamma Spectrometry:	2 inch by 2 inch NaI (T1) detector with 400 channel analyzer.
9. Calorimetry:	Thermal gradient calorimeter with thermopile output having a 223 micro volts/watt sensitivity at 60 watts, electrical heater calibration before and after each measurement.

Table III-2

Promethium-147 Encapsulation Test Results

Test	Test Weld	Inner Capsule Before Loading	Inner Capsule After Loading	Outer Capsule Before Loading	Outer Capsule After Loading	Test Weld
Radioactive Contamination Smear Tests	N.A.*	N.A.	< 1000 c/m prior to first weld; no smearable contamination background (< 300 c/m) after welding and decontamination	N.A.	no smearable contamination (background)	N.A.
Helium Leak Test	N.A.	—	no leaks	—	no leaks	N.A.
Dye Penetrant Test	no defects observed	no defects observed	underside defect in weld bead of EB closure	no defects observed	no indications on final closure	no defects observed
Ultrasonic Integrity Test	—	2 mil crack indications on ID	1-5 mil, 1-3 mil ID indication in seam weld end closures could not be tested	2 mil crack indications on ID	no indications, end closure could not be tested	—
Dimensions	N.A.	1.076" OD 0.031" wall 2.556" length	1.080" OD 2.565" length	1.131" OD 0.022" wall 2.641" length	1.135" OD 2.645" length	N.A.
Radiography	some porosity	some gas porosity	N.A.	some gas porosity	N.A.	some porosity
Metallography	some porosity	some gas porosity in seam weld	N.A.	some gas porosity in seam weld	N.A.	some porosity
Radiation Mapping	N.A.	N.A.	72 mr/hr at 0.5 m from capsule side; 45 mr/hr at 0.5 m from capsule end	N.A.	52 mr/hr at 0.5 m from capsule side	N.A.
Gamma Spectrometry	N.A.	N.A.	—	N.A.	Eu-154 and Pm-146 as reported in radio-chemical analysis	N.A.
Calorimetry	N.A.	N.A.	—	N.A.	59.3 ±0.2 thermal watts on 10/28/66	N.A.

*N.A. = not applicable

The promethium used for this capsule was taken predominately from two separation runs completed in September 1966. The radiochemical analysis, corrected to October 28, 1966, indicated the promethium oxide (Pm_2O_3) fuel contained:

Pm-147	164,000 curies
Pm-146	0.036 curies (Pm-146/Pm-147 = 2.2×10^{-7})
Pm-148m	1×10^{-4} curies (Pm-148/Pm-147 = 7×10^{-9})
Eu-154	41×10^{-4} curies (Eu-154/Pm-147 = 2.5×10^{-8})

Impurities detected spectrographically in the oxide included 3 percent of samarium-147 and less than 1 percent total of aluminum, lead, yttrium, and silicon.

a. Preparation of Pm_2O_3 Pellets. — Ten Pm_2O_3 pellets were pressed, sintered, and centerless ground to fit the inner liner of which nine were selected for the final load. The nine pellets provided a stack length of 2.440 inches and a total weight of 214.904 grams of Pm_2O_3. Average pellet density was 6.75 g/cc (90.9 percent theoretical density, based on 7.43 g/cc = 100 percent theoretical density (Reference 6). Pellet densities varied from 88.8 to 93.7 percent theoretical density.

Pellets were pressed at 126,000 psi in a special split sleeve die (Figure III-8) designed to yield 1.010 inch diameter pellets when pressed to 69 percent theoretical density and sintered to 90 percent theoretical density.

Sintering conditions* used for the Pm_2O_3 pellets loaded into the capsule were as follows:

Sintering Cycle	Furnace Temp., °C	Hours at Temp.
1	1355	45
2	1355	20
3	1415	26
4	1450	24

*The sintering of Pm_2O_3 had not been attempted prior to this program. Initial temperatures and times were chosen on the basis of studies using Sm_2O_3 as a stand-in.

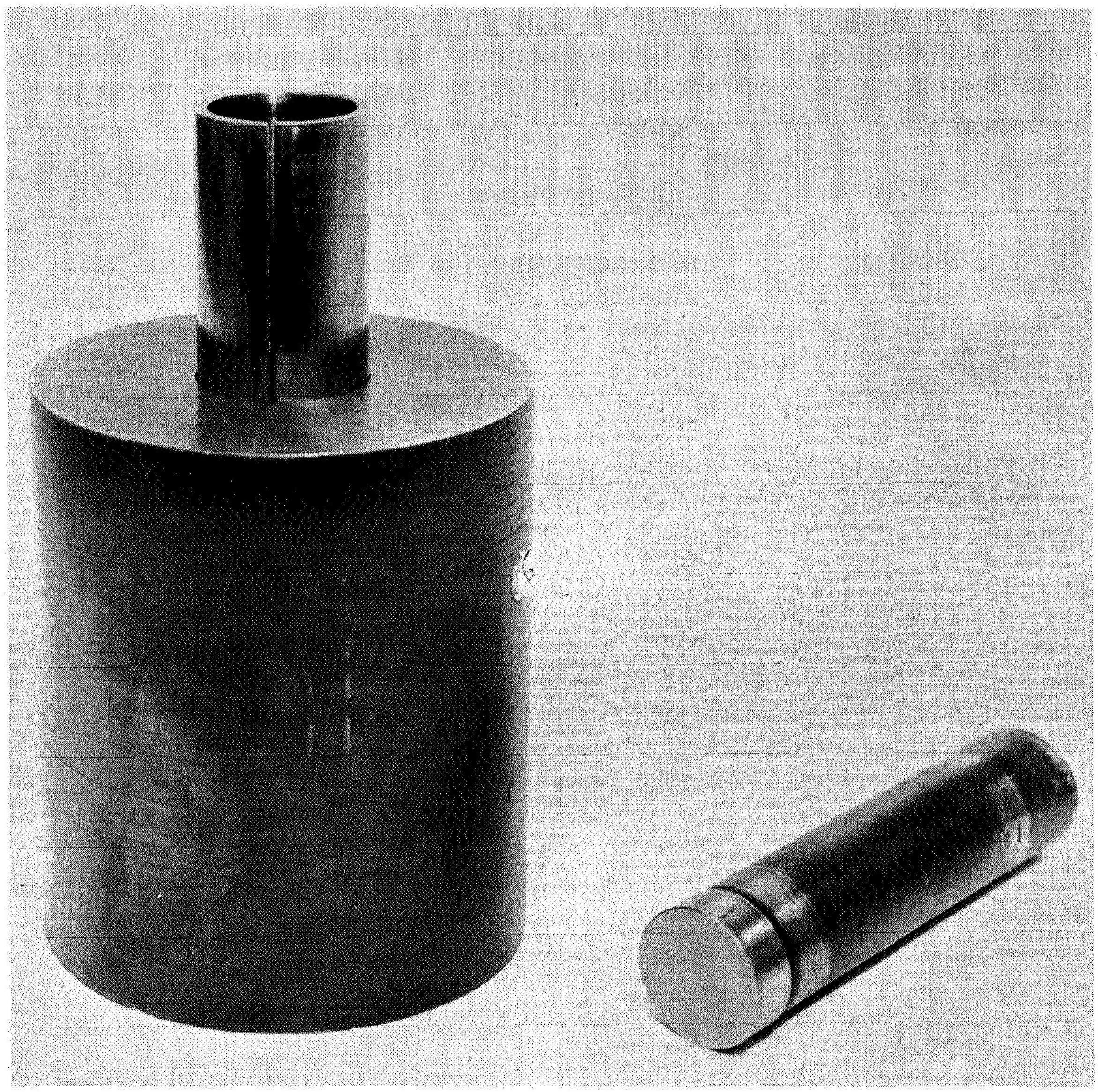


Figure III-8. Split Sleeve Die

Dimensions and densities of the pellets after the first sintering cycle and after the final sintering cycle are shown in Table III-3.

Because the pellets were too large in diameter to fit into the 1.014 inch ID inner capsule, a belt sander and rotating holder were devised to grind the pellets to fit the inner capsule. Two ring gages were used to determine when the pellets had been ground to the required diameter. A 1.015 inch ID gage gave a preliminary

Table III-3

Pm_2O_3 Sintering Data

Pellet	Wt., Grams	First Sintering				Final Sintering			
		Dia., In	Length, In.	Density		Dia., In.	Length, In.	Density	
				g/cc	% TD*			g/cc	% TD*
1	24.693	1.042	.269	6.569	88.4	1.035	.269	6.658	89.6
2	24.758	1.041	.271	6.550	88.2	1.033	.271	6.652	89.5
3	24.711	1.038	.269	6.625	89.2	1.033	.269	6.689	90.0
4	24.796	1.029	.267	6.815	91.7	1.022	.267	6.909	93.0
5	24.759	1.026	.267	6.845	92.1	1.021	.265	6.964	93.7
6	24.676	1.038	.269	6.615	89.0	1.032	.270	6.668	89.7
7	24.757	1.024	.270	6.794	91.4	1.019	.271	6.836	92.0
8	24.708	1.045	.272	6.463	87.0	1.036	.271	6.600	88.8
9	24.735	1.028	.267	6.811	91.7	1.024	.267	6.865	92.4

*TD – theoretical density, 7.43 g/cc = 100 percent

indication that the pellet was nearly the correct size. When a 1.010 inch ID gage could be passed over the pellet, grinding was stopped.

Nine Pm_2O_3 pellets were loaded into the inner capsule liner in a hot cell (Figure III-7). Prior to loading, each pellet was brushed to remove traces of loose Pm_2O_3 or powder picked up during grinding. A 0.003 inch thick foil of the same material as the inner cladding was pressed into the capsule after all of the pellets were loaded, to facilitate decontamination and welding. The inner capsule was wrapped in aluminum foil before loading and removed from the foil immediately before it was brought out of the hot cell. The capsule was decontaminated easily by wiping with cotton-tipped swabs moistened with water. A lid was placed on the decontaminated capsule, the capsule placed in a chill block, and the assembly transferred to the welding box. The final weight of the loaded inner capsule was 320.079 grams.

b. Calorimetry. — The thermal output of the doubly clad fueled capsule was determined in an available thermal-gradient calorimeter having a thermo-pile response (at 60 watts) of 223 microvolts per watt. Calibration was by substituted electrical heater (voltage and current measurements). The calorimeter was calibrated at power levels bracketing the capsule, and the thermal power was measured at 59.3 ± 0.2 watts (on 28 October 1966) expressed at the 2-sigma confidence level.

Figure III-9 illustrates in semi-log coordinates the actual thermal decay of the thruster fuel capsule. Note that the curve represents the decay of promethium-147, whereas the thermal power of the capsule includes the activities of Pm-146, Pm-148m, and Eu-154, each with their own decay characteristics. However, the small amounts of these contaminants (cf. Section III-B-2, "Promethium-147 Fuel") will not affect the curve of Figure III-9 to any noticeable degree. For example, the thermal activity of all the contaminants combined will not amount to the 0.2 watts of thermal power tolerance discussed above.

3. Electrical Simulator

The purpose of the electrical simulator is to approximate the thermal and outer physical characteristics of the promethium-147 fueled heat source. The thruster testing approach was to initially test the TSK 2000-1RE assembly with the electrical heater, thereby precluding any radiation hazards associated with the Pm-147 heat source, and then continue with the fueled thruster tests after facility and instrumentation checkout.

The electrical heater, shown in the radiograph Figure III-6, is a 0.25 inch diameter magnesium oxide rod around which is wound a 0.01 inch diameter

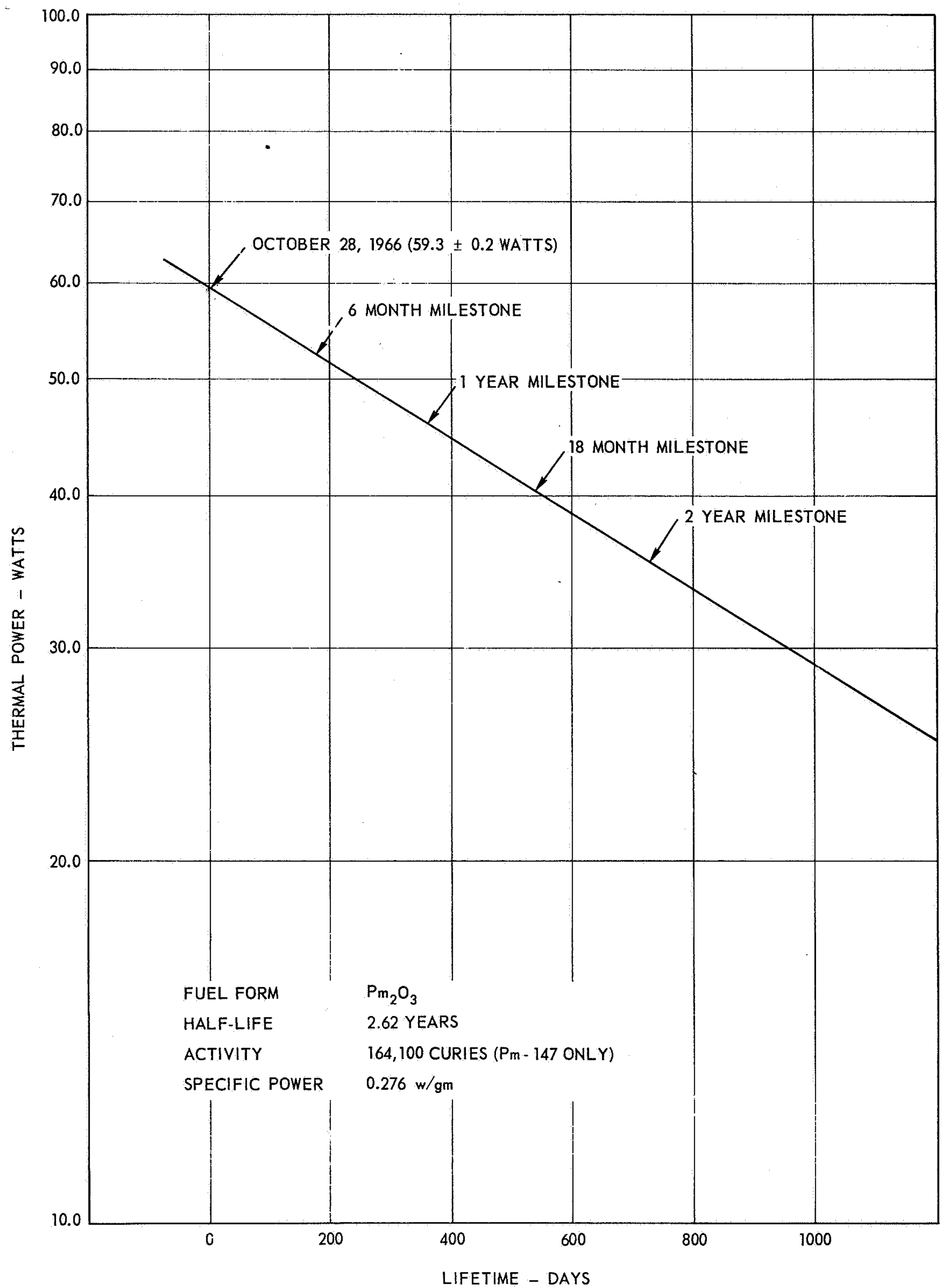


Figure III–9. Pm_2O_3 Fuel Capsule Thermal Decay History

platinum-rhodium heater wire. The wire is separated from the Hastelloy-X sheath by a thin fill of powdered boron nitride. After being compacted during a swaging operation, the temperature differential between the wire and sheath when heated to 2000°F has been measured at approximately 50°F in a typical thruster application. A thermocouple, type K (chromel-alumel), is assembled into the heater core prior to swaging and provides a measurement of maximum heater temperature.

C. RADIATION DOSE RATES

The dose rates of the fueled thruster (including heat shield) were measured at twelve inches from the surface contour. These measurements were made following the tests at Mound Laboratory and the RIJ's return to Battelle Northwest in January 1967. The measurements were made at nine different points around the thruster assembly. Figure III-10 illustrates the dose rates at the measured twelve inch values and also reflects the anticipated (calculated) dose rates at 1 meter* (39 inches). Each measurement was repeated at 60° intervals as the device was rotated around the cylindrical axis. No significant variations were observed in the repetitive measurements and the values shown (at twelve inches) in Figure III-10 are the average of six revolutions. The classified Appendix (D) elaborates on the thruster shielding and radiation properties.

Note, in Figure III-10, the dose rate at the 50 centimeter (0.5 meter) point. Both the calculated and the measured values are indicated. The agreement is somewhat better than expected considering the 5% uncertainty in the impurity content and a similar uncertainty in the decay schemes.

*Shipping regulations require that the radiation dose rate shall not exceed 10 mr/hr at 1 meter from the container surface.

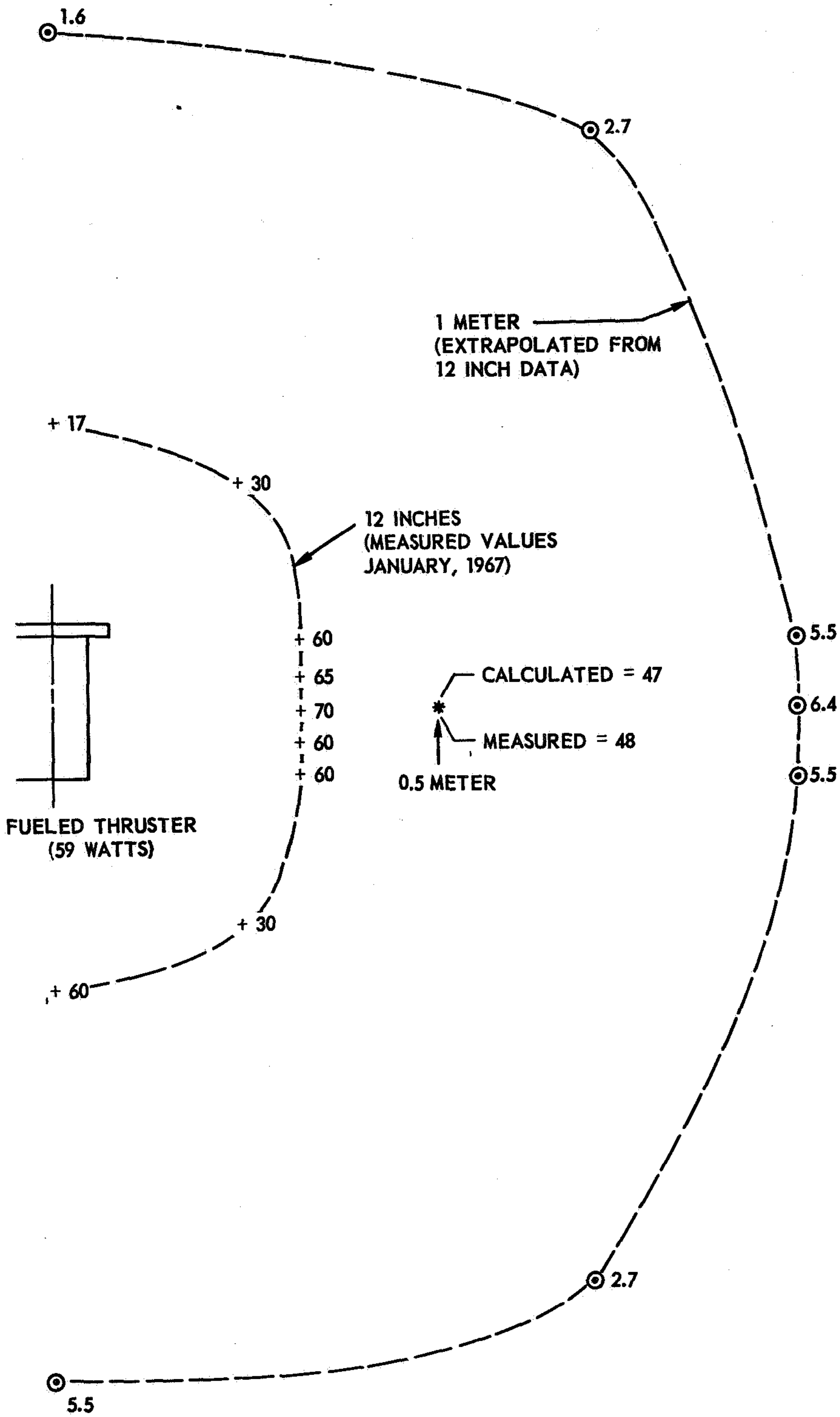


Figure III–10. Radiation Dose Rates (mr/hr) from Thruster Surface

IV. TEST FACILITIES AND INSTRUMENTATION

A. GENERAL REQUIREMENTS

Thruster tests were to be performed over a range of rated thrust levels from 0.0 lbs to 0.100 lbs at propellant mass flow rates ranging from 6×10^{-5} lbs per second to 2×10^{-4} lbs per second. Vacuum facility requirements were based upon a capability for maintaining an ultimate pressure of 1×10^{-5} torr or less when the chamber is clean, dry, and empty. Vacuum working pressure permitted an ammonia flow rate of 1×10^{-4} lbs per second while maintaining 40×10^{-3} torr or less.

For design purposes, the ammonia was considered completely decomposed into nitrogen and hydrogen. The required pumping capacity, determined on the basis of completely dissociated flow, was checked to assure that equal mass flow rates of undissociated ammonia could also be handled. It was also assumed that propellant flow would be maintained for a period sufficiently long to permit establishment of steady state conditions within the thruster.

Test instrumentation included power supplies, signal generator, sensors and appropriate readout equipment for measurement of:

(1) Propellant supply and thruster chamber pressure,

(2) Propellant and thruster body temperature profiles,

(3) Vacuum system pressure level,

(4) Thrust,

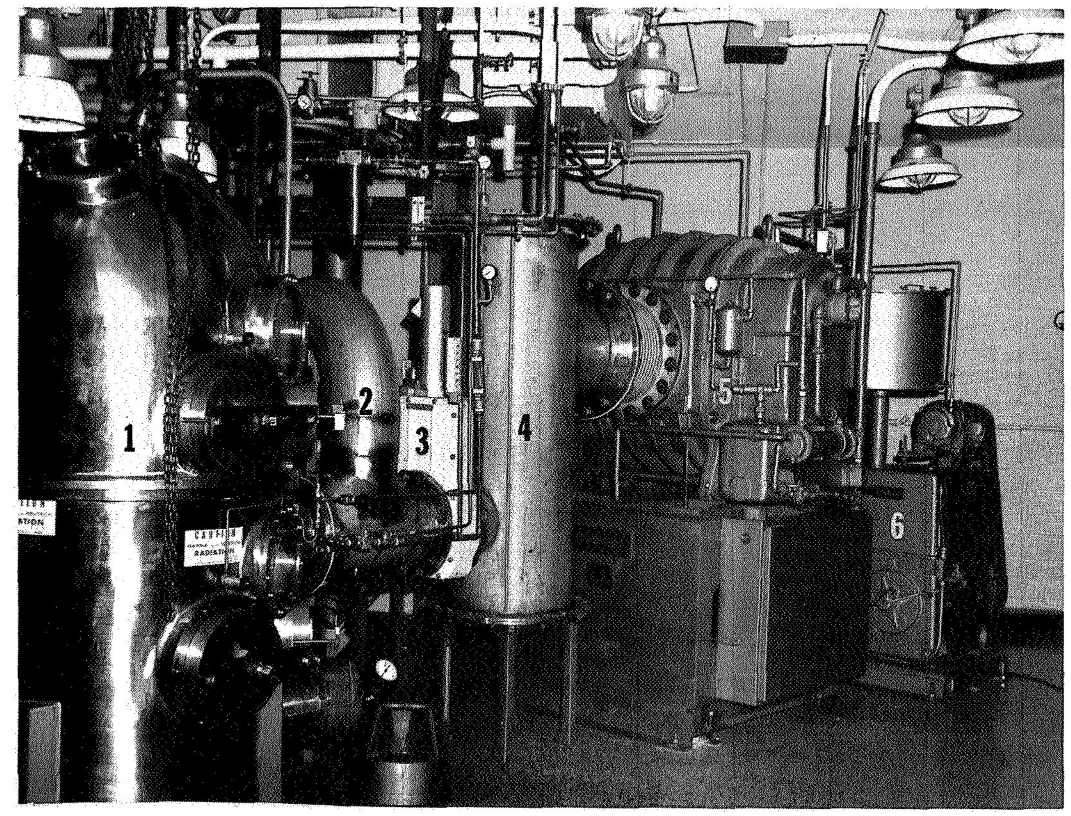
(5) Current,

(6) Voltage.

Two ranges of vacuum system pressure levels were required. The upper range covered from 1 to 1000 microns. The lower range covered from 1 micron down to ultimate vacuum pressure.

B. THRUSTER TEST FACILITIES

The promethium-147 fueled thruster hot tests and some electrically heated (for facility checkout) tests were conducted at Mound Laboratory, Miamisburg,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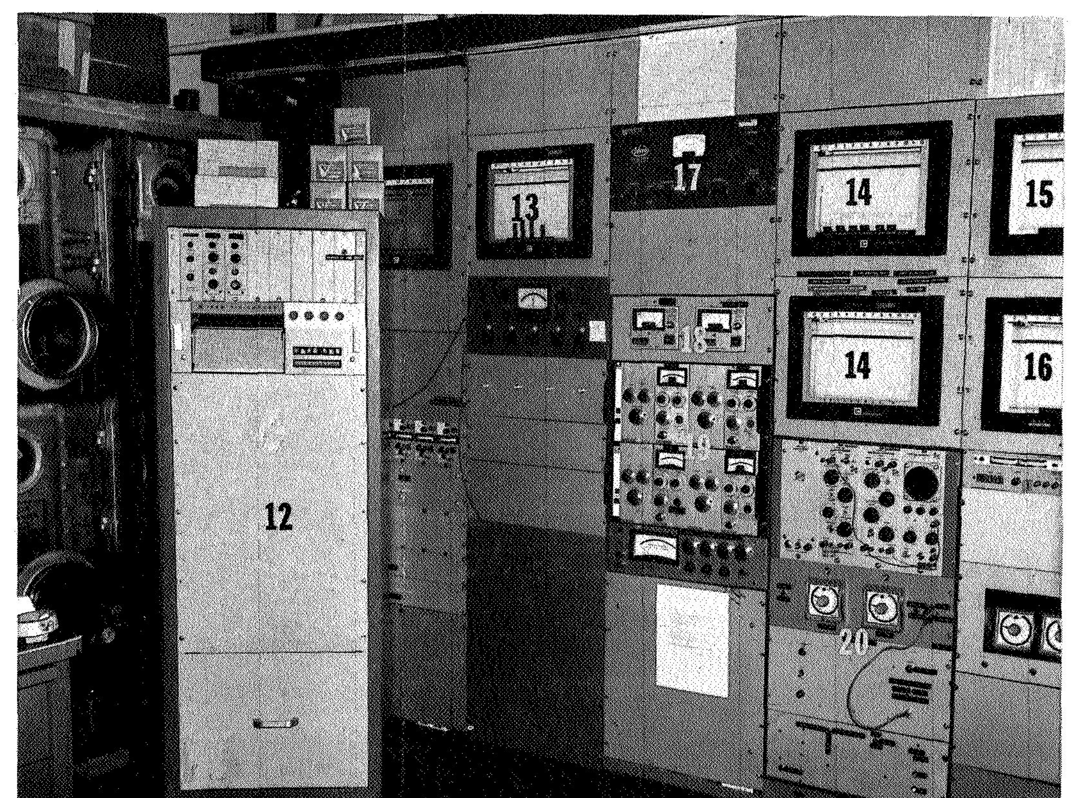
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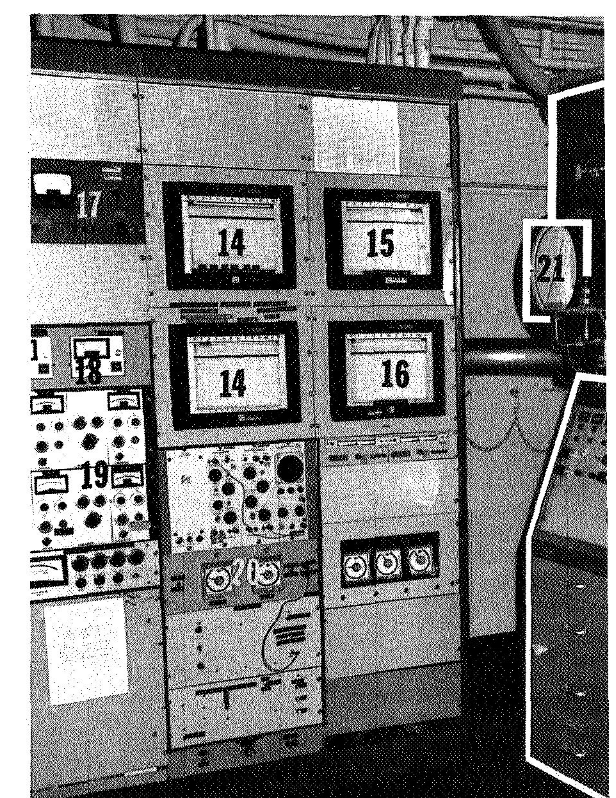
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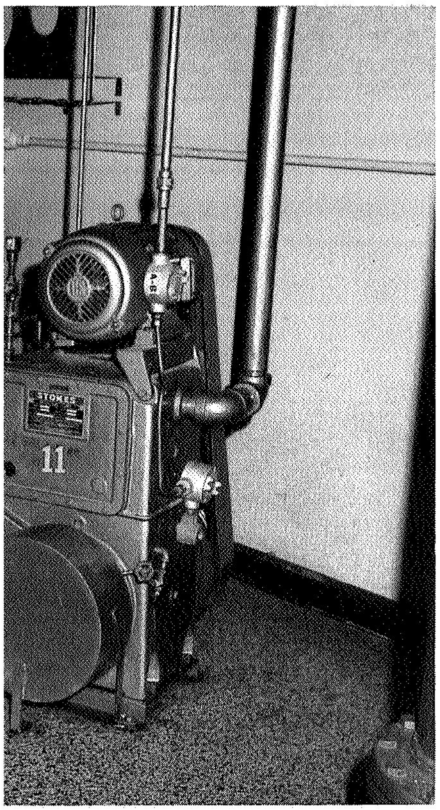
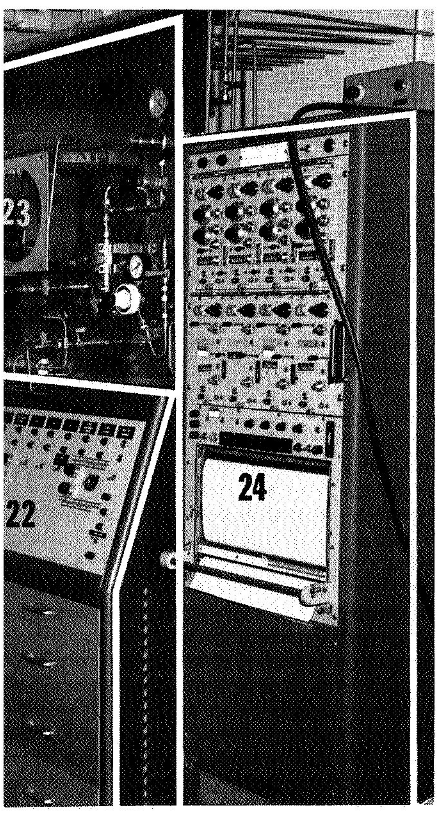




LEGEND

Item Number	Description
1	Radioisojet Vacuum Test Chamber
2	10-Inch vacuum line to diffusion pump
3	10-Inch gate valve (EP-2) to stokes 1719 vacuum system
4	20-Inch vacuum manifold
5	Blower section of stokes 1719 vacuum system
6	Mechanical pumps (2 each) of stokes 1719 vacuum system
7	10-Inch gate valve (EP-1) to diffusion pump
8	Baffle
9	10-Inch CVC type PMCS-10 diffusion pump
10	Refrigeration unit for baffle
11	Stokes 140 cfm roughing pump
12	Minneapolis-Honeywell type 1508 Visicorder
13	L & N Speed-O-Max Recorder (voltage & current for electrical microthruster test)
14	L & N Speed-O-Max Recorder (temperature)
15	L & N Speed-O-Max Recorder (vacuum)
16	L & N Speed-O-Max Recorder (radiation level)
17	Veeco RG-2A vacuum gauge
18	Veeco TG-6 vacuum gauges
19	RIDL 35-7B radiation monitors
20	Timers, Tektronix Type 161-2 and Eagle Signal Type Cycl-Flex
21	Pressure gauge, Heise, Model C, 0-400 p.s.i.
22	Control console panel
23	Ammonia and helium pressure regulators
24	Sanborn Model 350 recorder

Figure IV-1. Mound Laboratory RIJ Test Facility

2

Ohio. The facility at Mound, modified specifically for the microthruster test program, consisted basically of two rooms (T-61) illustrated in the sequence of photographs in Figure IV-1. Figure IV-1a and b (items numbers 1-11 of the legend) show the primary test facility consisting of the vacuum chamber and pumps. The second room, Figure IV-1c and d, contained all of the control panels, readout equipment, and gauges. Detail flow charts for water, electricity, and process are included in Appendix A, in addition to a detail of the control panel.

Figure IV-2 is a schematic of the displacement type thrust measurement system employed during the TSK 2000-1RE performance tests. Figure IV-3 is a photograph of an identical system installed in a General Electric Company facility. The thrust rig was designed to provide approximately one mil displacement per millipound of thrust.

C. INSTRUMENTATION

Thrust, as measured by the amount of thrust rig table displacement, was sensed by a Daytronic Model DF 160 Linear Displacement Transducer, mounted on a rigid portion of the thrust rig. The system was calibrated as outlined in Instruction I of the Standard Operating Procedures (Appendix B), so that the output could be read directly in millipounds on a Sanborn strip chart recorder. The calibration weight pan was allowed to remain in place during the tests to provide some additional damping of table oscillations during thruster pulsing.

Flow control orifice pressure and thruster chamber pressure were measured with a Pace Model P7D-100 (0-100 PSID) differential pressure transducer mounted on the thrust rig table. A Hoke 3-way solenoid valve was used to read selectively the pressure at either of these locations. The pressure transducer was calibrated by reading the line pressure at no-flow conditions on a Heise Pressure Gauge and setting the orifice pressure readout correspondingly, thereby using the Heise gauge as a standard. This calibration could be checked and reset if necessary whenever the thruster valve was "off". Orifice pressure was measured as line pressure at the thrust rig table. The chamber pressure tap, a fitting in the thruster itself, actually measured internal pressure which was essentially nozzle inlet pressure (reference Figure IV-4 for approximate pressure tap locations).

All temperature measurements were made with Type K (Chromel-Alumel) thermocouples spot welded to the thruster in accordance with the sketch of Figure IV-4. The core temperature thermocouple for the electrically heated thruster was built into the heater cartridge as described in Section III-B-3.

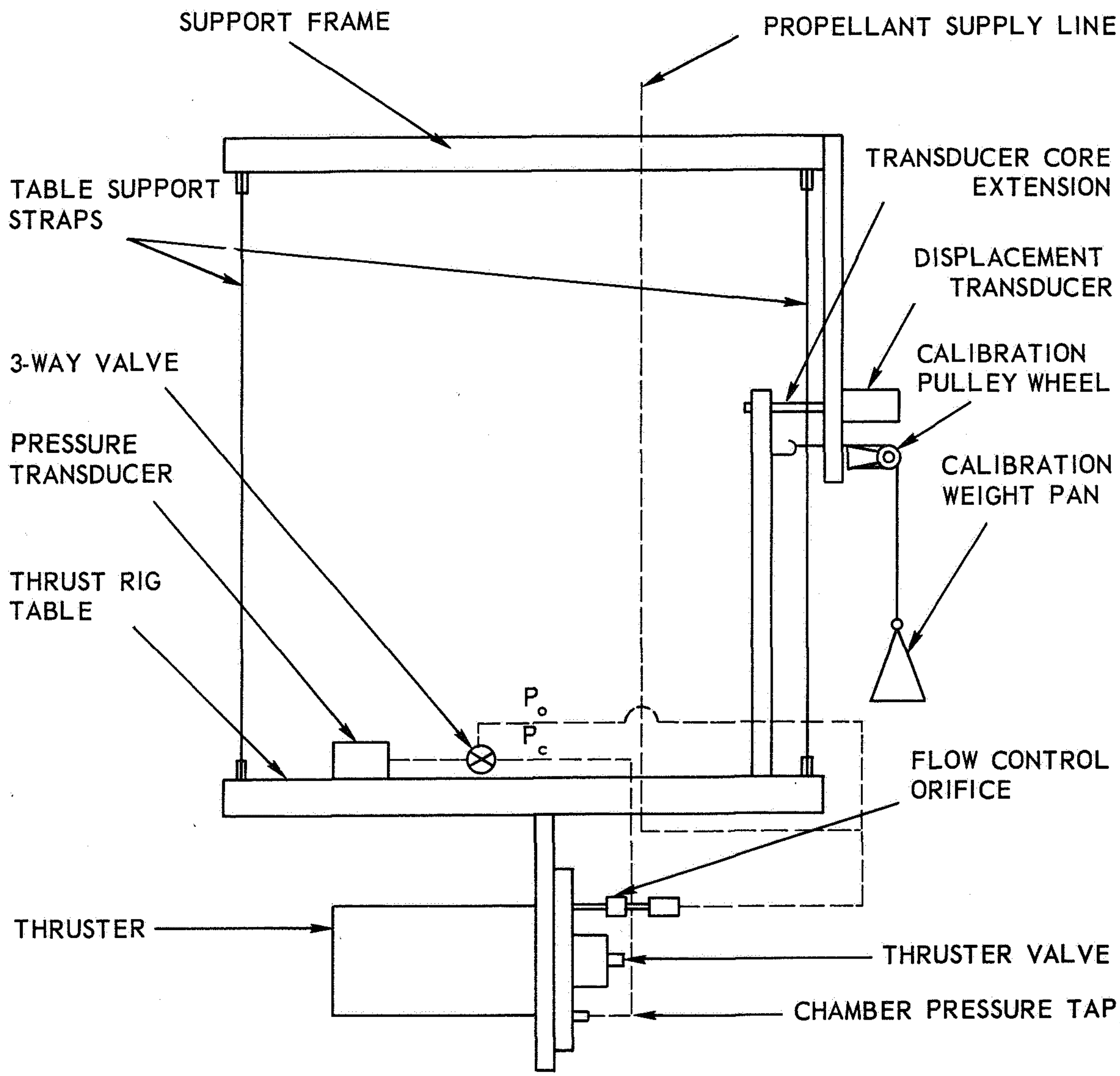


Figure IV–2. Schematic: Thrust Measurement System

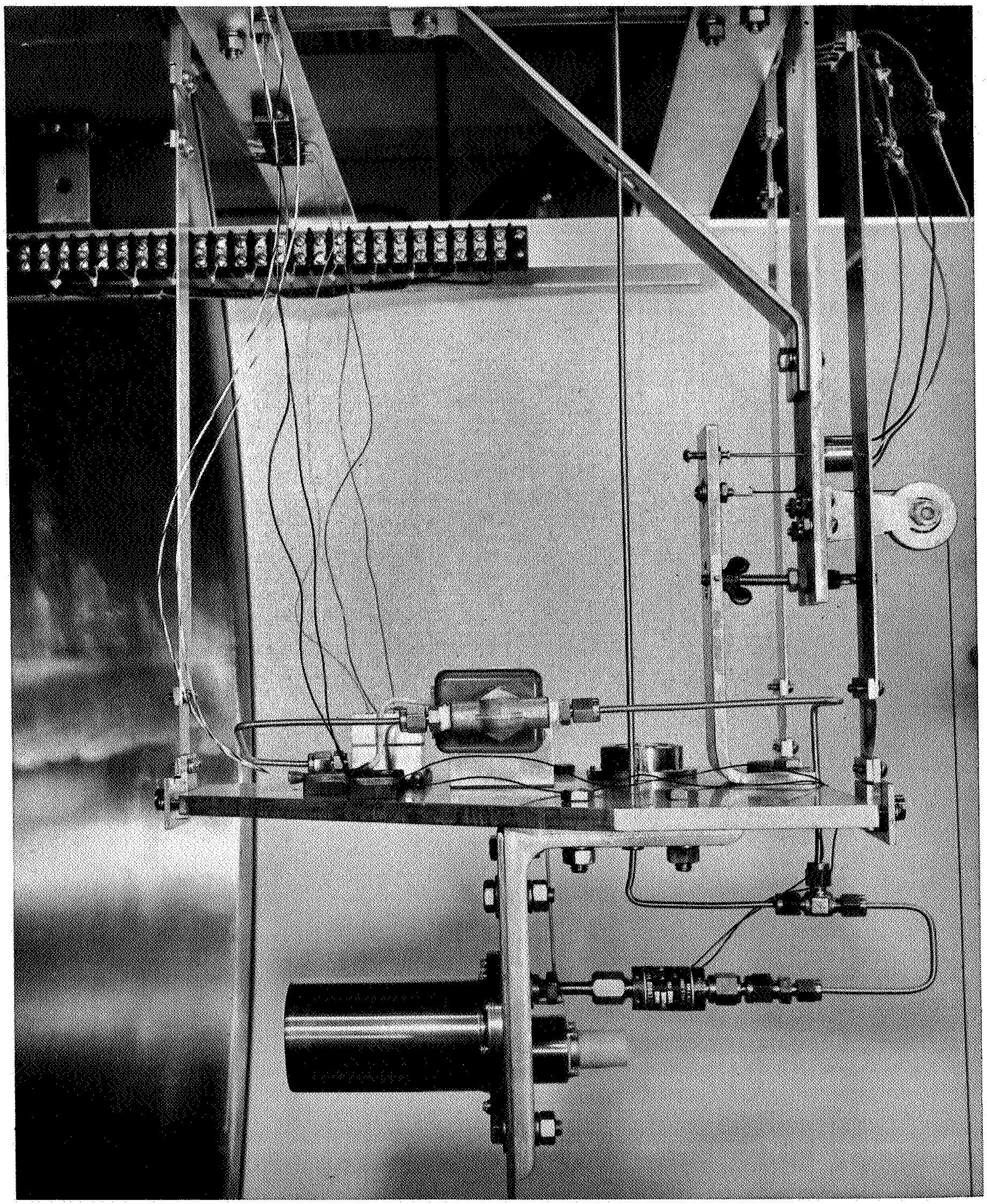
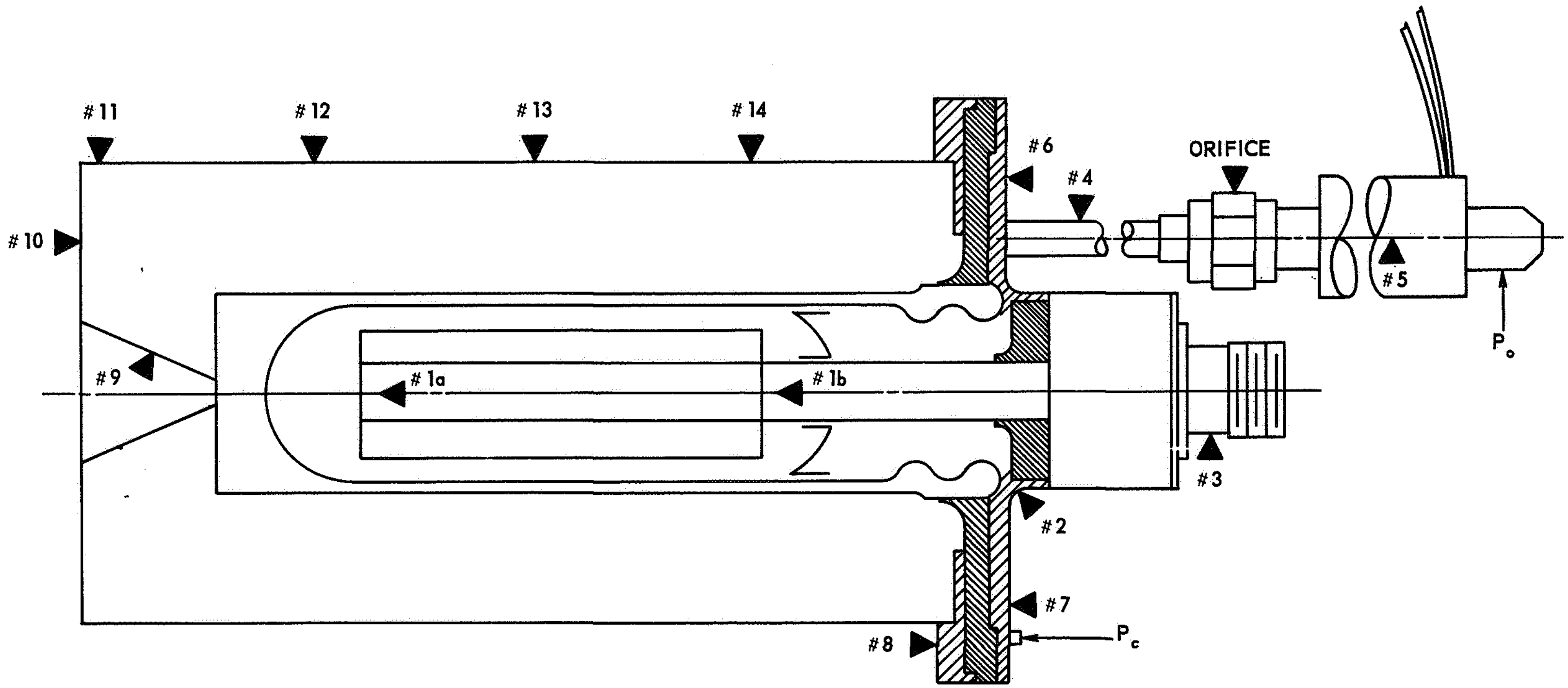


Figure IV–3. Photo: Thrust Measurement System



1a. ELECTRICAL HEATER CORE T/C
1b. Pm-147 CAPSULE T/C

Figure IV–4. Thermocouple Locations–TSK 2000-1RE

The core temperature thermocouple for the promethium fueled thruster was a 1/16 inch diameter, stainless steel sheathed, dual thermocouple inserted into the end of the capsule housing to within approximately 1/16 inch of the capsule cladding. One element was read on a 12 point temperature compensated recorder and the second was fed to a millivolt potentiometer through a 32°F reference junction. The outputs of these two thermocouples were periodically compared and found to agree within one percent. All other thermocouples were read on 12 point L and N Speed-O-Max recorders.

V. TEST RESULTS

A. TEST PROCEDURE

The electrically heated thruster was tested prior to the promethium-147 fueled unit to establish a basis for comparison of fueled thruster performance. The preliminary testing also served as a means of assuring that all instrumentation and support equipment functioned properly. Additional electrically heated thruster tests were performed subsequent to the radioisotope fueled tests to check for heat shield degradation or other system changes which may not have been detected during the fueled tests.

Briefly stated the test program consisted of the following sequence:

Thruster	Location	Purpose	Remarks
Electrical	GE—Evendale	Initial Thruster System Calibration	Define general procedures
Electrical	Mound Lab—Miamisburg	Instrumentation & Facility Checkout	System shake-down and to obtain data for comparison with RIJ test data
Radioisotope	Mound Lab	To Obtain RIJ Data	(cf. RIJ Program Objectives Section II-B)
Electrical	Mound Lab	Post Test Calibration	Verify heat shield performance
Electrical	GE—Evendale	Final Verification Data	Additionally, the fueled inner thruster body (Pm-147 capsule plus Hastelloy-X sheath) was returned to BNW where radiation mapping was conducted (cf. Section III-C).

The thruster unit assembled at GE for the verification tests consisted of (1) the heat shield assembly used for all testing at Mound Laboratory, (2) the outer thruster body and nozzle used for the fueled testing, and (3) the electrically heated inner thruster body used at Mound Laboratory. This assembly would allow use of the same thruster nozzle for electrical testing as was used

for the fueled testing rather than using a similar nozzle as was previously done at Mound. Power vs Temperature data obtained during the four electrical thruster tests are shown in Figure V-1. Discussion of these tests and the isotopic fueled thruster tests follows.

B. DISCUSSION OF TEST RESULTS

1. Electrically Heated Thruster

The first tests of the electrically heated TSK 2000-1RE thruster were performed at the GE Propulsion Laboratory in a 10^{-8} torr atmosphere. Under these conditions, as shown in Figure V-1, the design power of 60 thermal watts resulted in a measured heater core temperature of 1860°F. Subsequently, at Mound Laboratories under similar conditions (except at a poorer vacuum level, 10^{-5} torr), the same power input provided a measured core temperature of 1740°F. After approximately 4 weeks of testing under the Mound Lab vacuum conditions, the same thruster system was retested; at 60 watts, the temperature measured in the same location was only 1670°F. As can be seen from the curve of Figure V-1, the data obtained during each of these tests were quite consistent, indicating that there was a degradation in thruster core temperature.

The thruster was returned to the GE Propulsion Laboratory and retested following the fueled thruster tests at Mound. The first temperature point (at 52 watts) corresponded well with the curve generated from the final data collected at Mound Laboratories. Subsequently, after more than a day at a vacuum of 10^{-7} to 10^{-8} torr, three additional data points obtained generated a curve displaced from the initial GE data curve by only 20°, i.e., at 60 watts the final GE test data indicated a core temperature of 1840°F as compared to 1860°F from the initial data. It is postulated that the apparent heat shield package performance degradation associated with the test sequence conducted at Mound Laboratories is imposed by test facility limitations and would not occur in space, i.e., the attainable vacuum in the Mound facility is not "hard" enough.

Representative temperatures are illustrated in the schematic of Figure V-2. The actual thermocouple locations are shown in Figure IV-4. The average of the measured temperatures, as shown in the schematic, were obtained during the initial tests at GE when the power input of 65 watts resulted in a measured core temperature of 1900°F.

After installing the electrically heated thruster in the thrust rig at Mound Laboratory, the vacuum chamber was evacuated to 1 micron. The upstream and

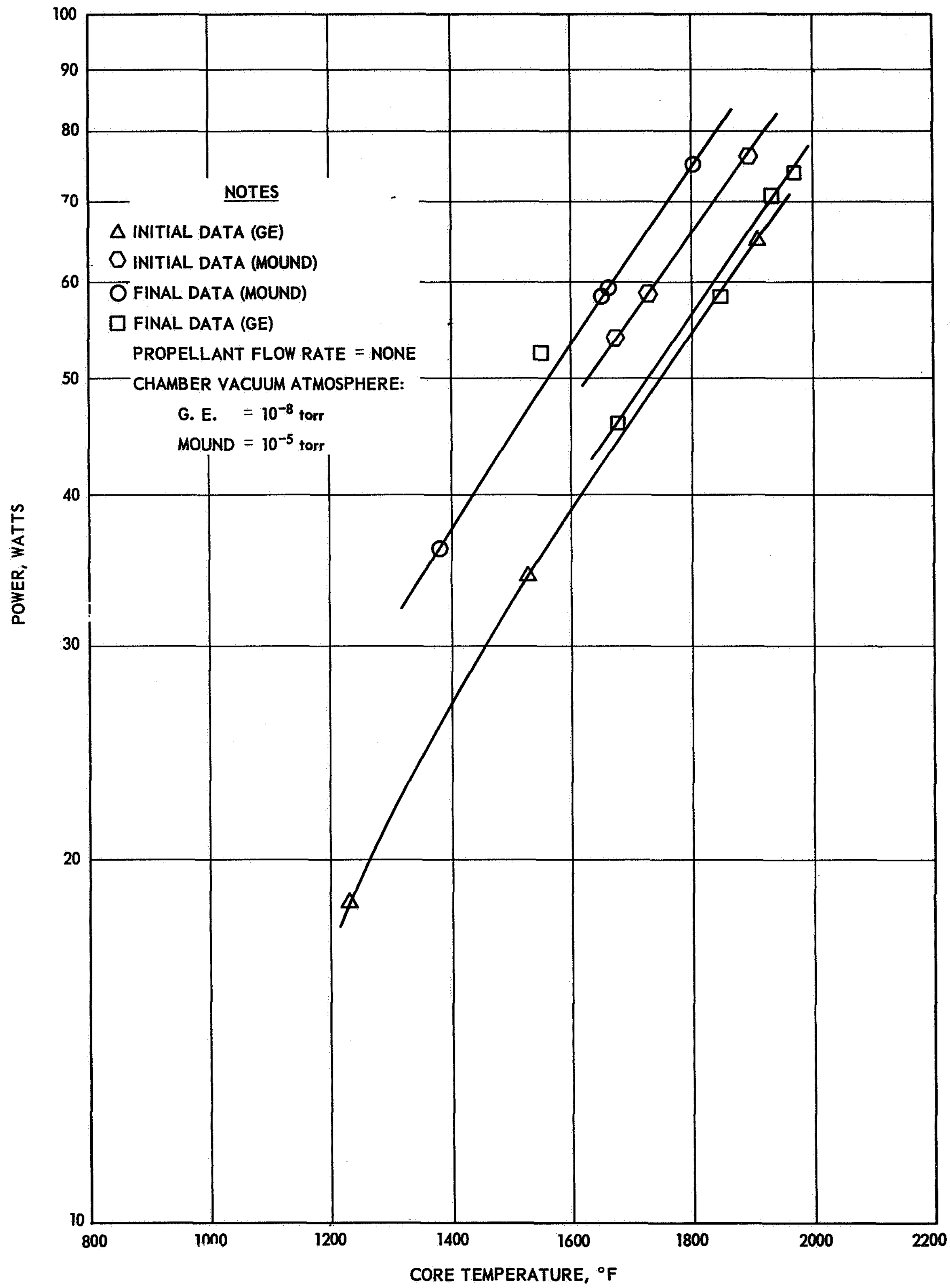


Figure V–1. Electrical Power vs Thruster Core Temperature

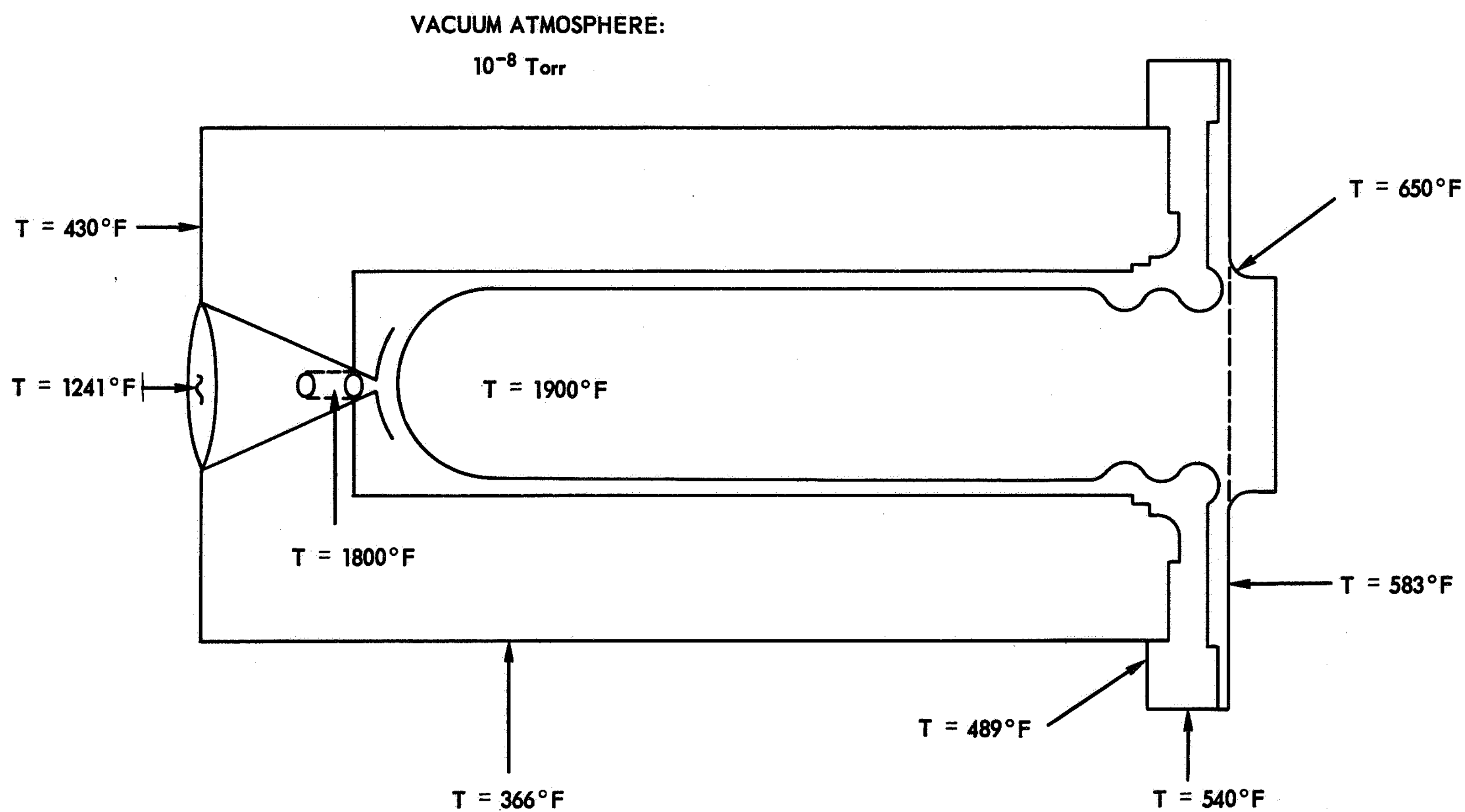


Figure V–2. Schematic: RIJ Measured Average Temperature Distribution

downstream orifice pressures and thrust data were recorded, with the core at room temperature, by pulsing both helium* and ammonia propellants for five to ten seconds. The downstream orifice pressure was assumed to approximate the thruster chamber pressure, P_c. The heater was energized to 59 watts (the expected isotope power level) which resulted in a stabilized core temperature of 1740°F as read by thermocouple number 1a. (cf. Figure IV-4). Pressure and thrust measurements were again taken using both propellants. These data, along with the cold (room temperature) data, are summarized in Figures V-3 (Thrust vs Chamber Pressure, P_c) and V-4 (Orifice Pressure, P_o, vs Chamber Pressure, P_c).

2. Promethium-147 Fueled Thruster

On receipt of the radioisotope fueled inner and outer thruster bodies, the electrical simulator (electrically heated inner and outer thruster bodies) was removed from the heat shield package which remained mounted in the thrust rig. The radioisotope fueled unit (inner and outer thruster bodies) was inserted into the heat shield as specified in the S.O.P. (Appendix B), and the test chamber evacuated. The temperature rise of the thruster core during chamber evacuation is shown in Figure V-5.

Under vacuum with all temperatures stabilized, the thruster was pulsed for five second intervals using helium and ammonia as propellants at chamber pressures from 5 to 25 psi. During these pulses, thrust (F_o), orifice pressure (P_o), and chamber pressure (P_c) measurements were made. These data and the calculated specific impulse** (I_{sp}), as plotted on Figures V-6, V-7, and V-8 were obtained from this initial test series. During the standard pulse testing runs, i.e., five seconds on -200 seconds off etc., data were obtained to define a plot of I_{sp} and indicated core temperature, Figure V-9. (Figure V-9 was generated from thrust and orifice pressure readings at a chamber pressure of one atmosphere of NH_3.)

3. Comparison of Electrical/Fueled Thruster

Of primary interest was the delivered specific impulse as calculated from measured data during the testing of both the electrically heated and the fueled RIJ's. The initial data obtained with the electrically heated thruster indicated specific impulses of greater than 250 seconds. Subsequently, during the promethium-147 tests, it was found that the electrical data were in fact

*Helium was used as a propellant standard and also as a back-fill gas.

**Specific Impulse = Thrust (lbs)/propellant flow rate (lbs/sec)

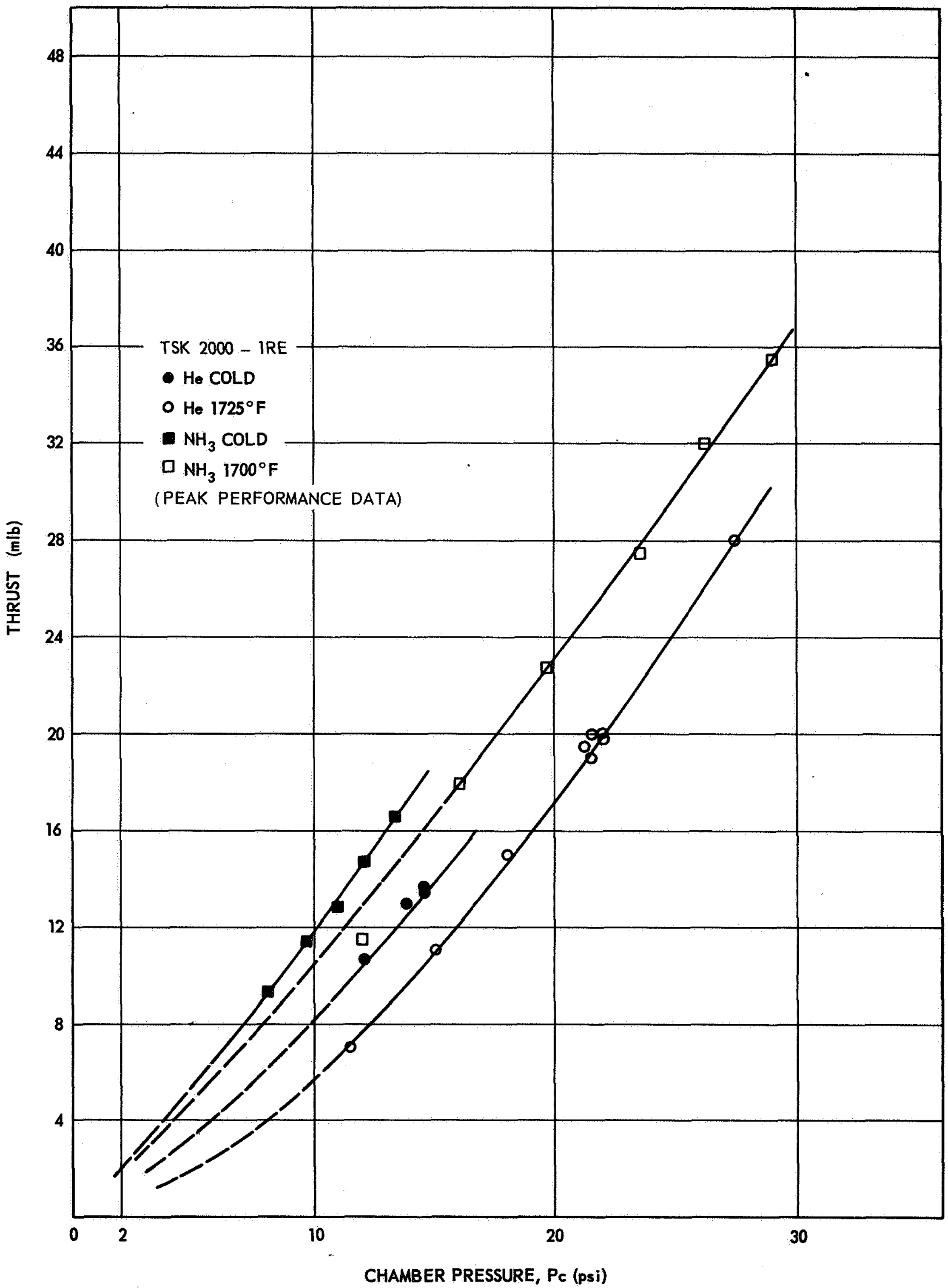


Figure V–3. Electrically Simulated Radioisojet, Thrust vs Chamber Pressure

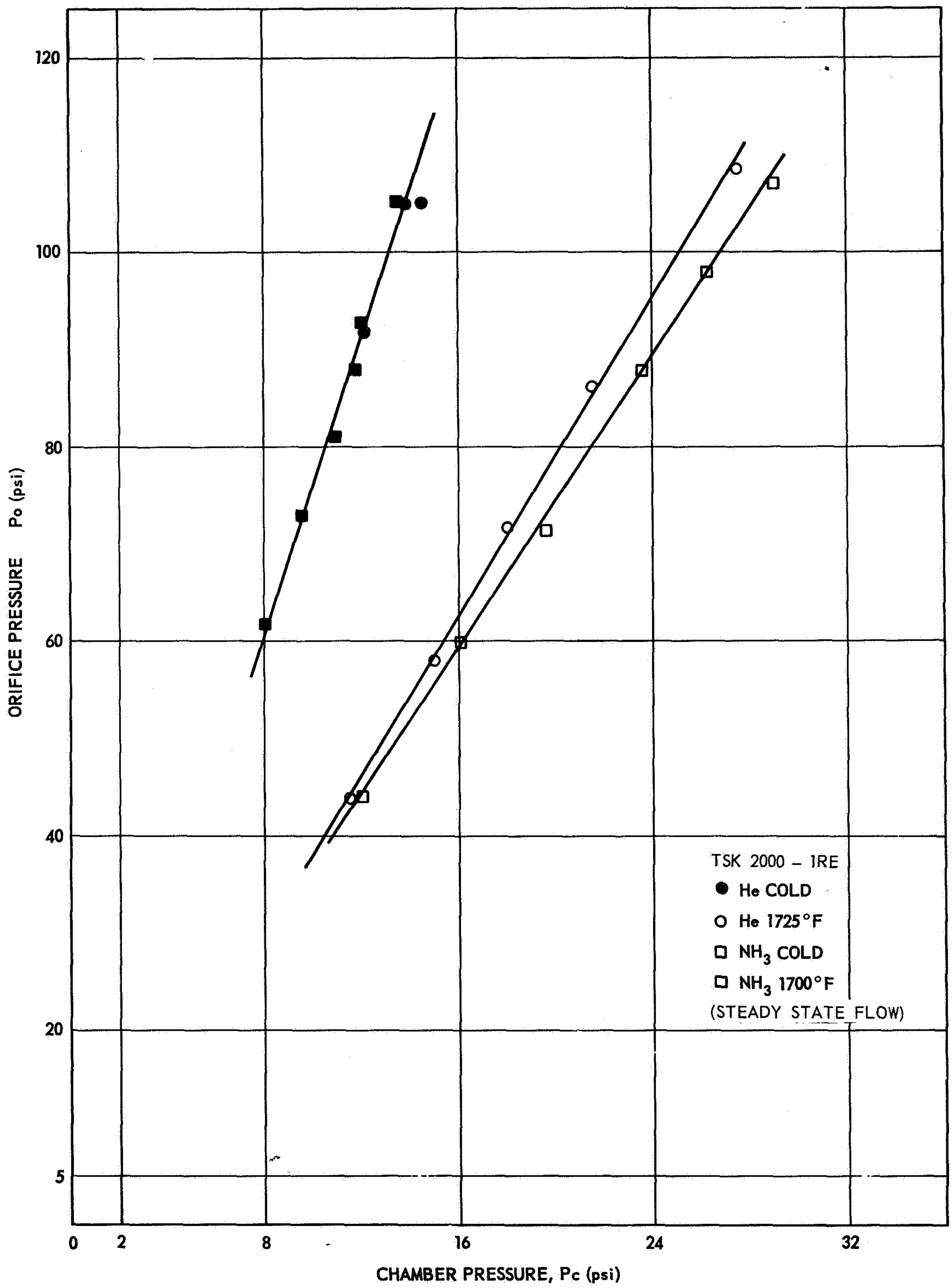


Figure V–4. Electrically Simulated Radioisojet, Orifice Pressure vs Chamber Pressure

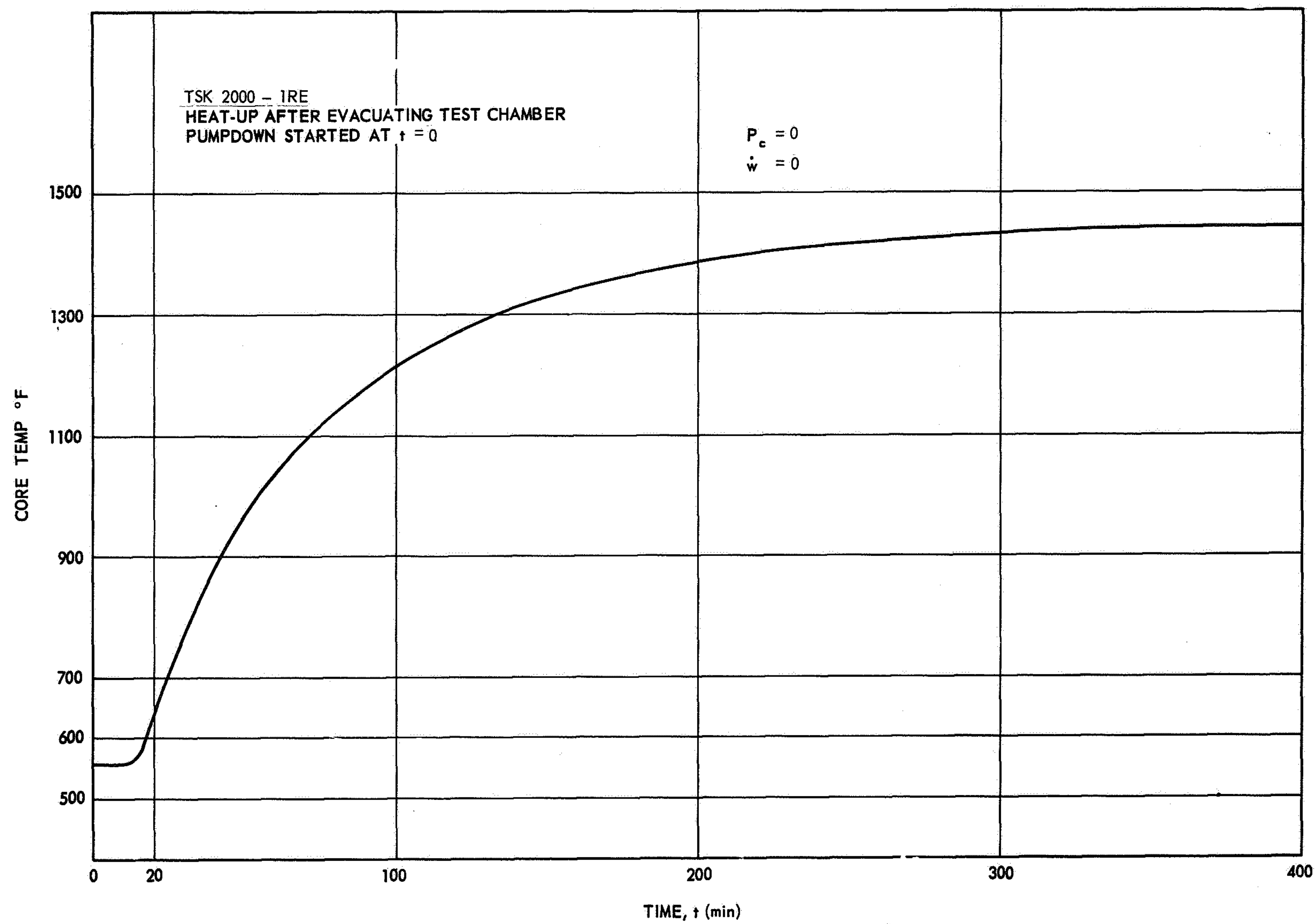


Figure V–5. Promethium-147 Fueled Thruster, Temperature vs Time

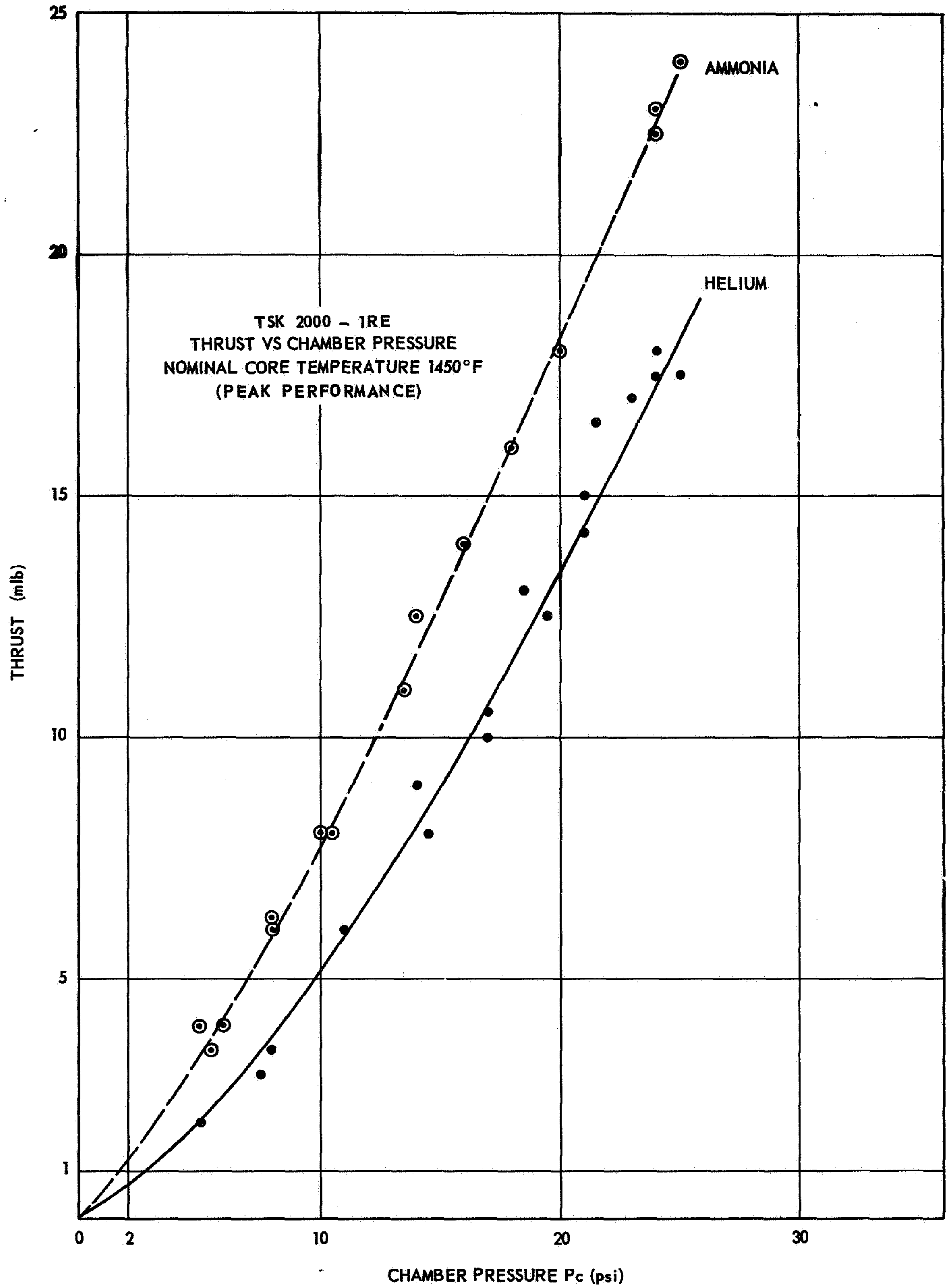


Figure V–6. Promethium-147 Fueled Thruster, Thrust vs Chamber Pressure

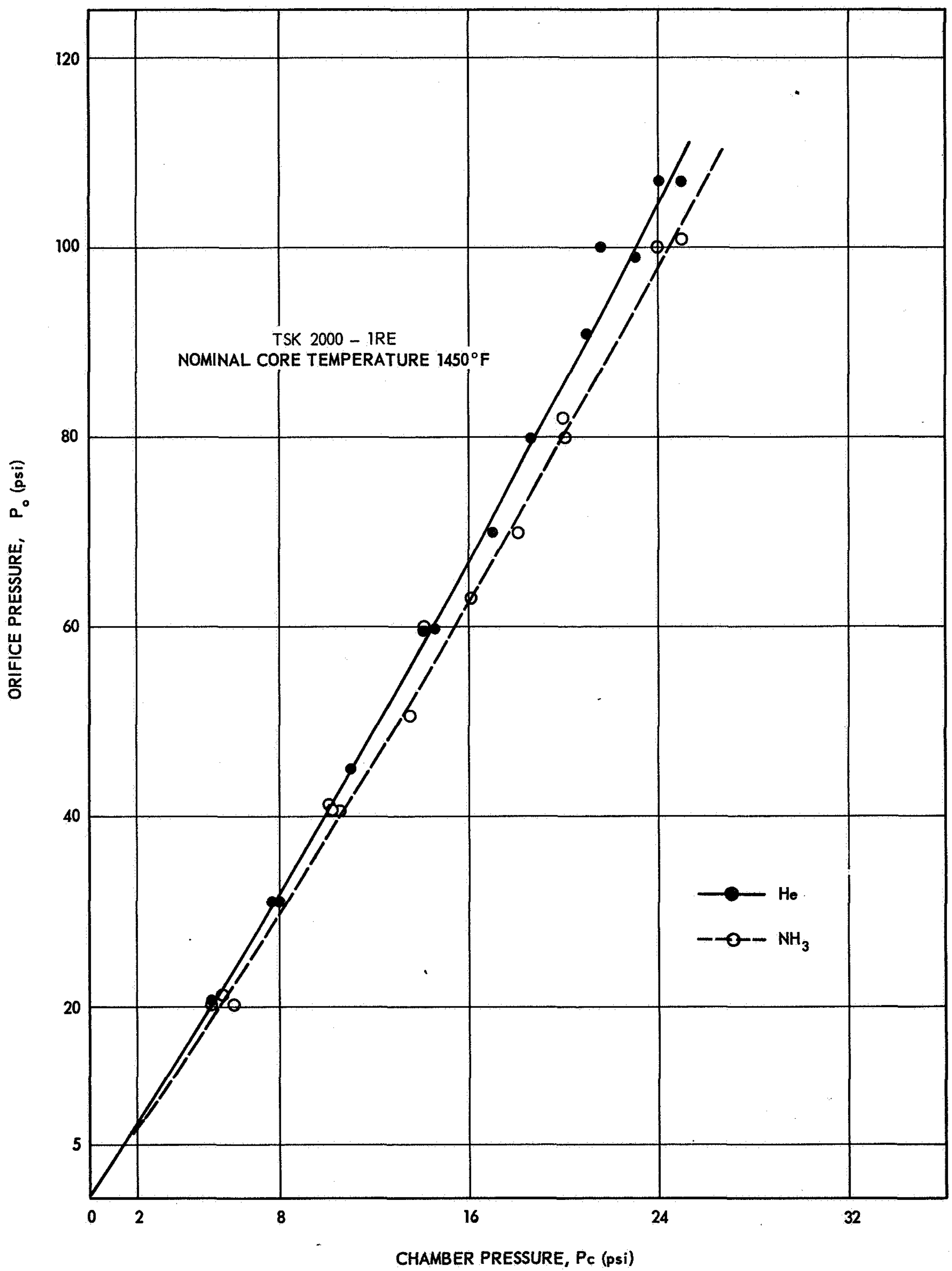


Figure V–7. Promethium-147 Fueled Thruster, Orifice Pressure vs Chamber Pressure

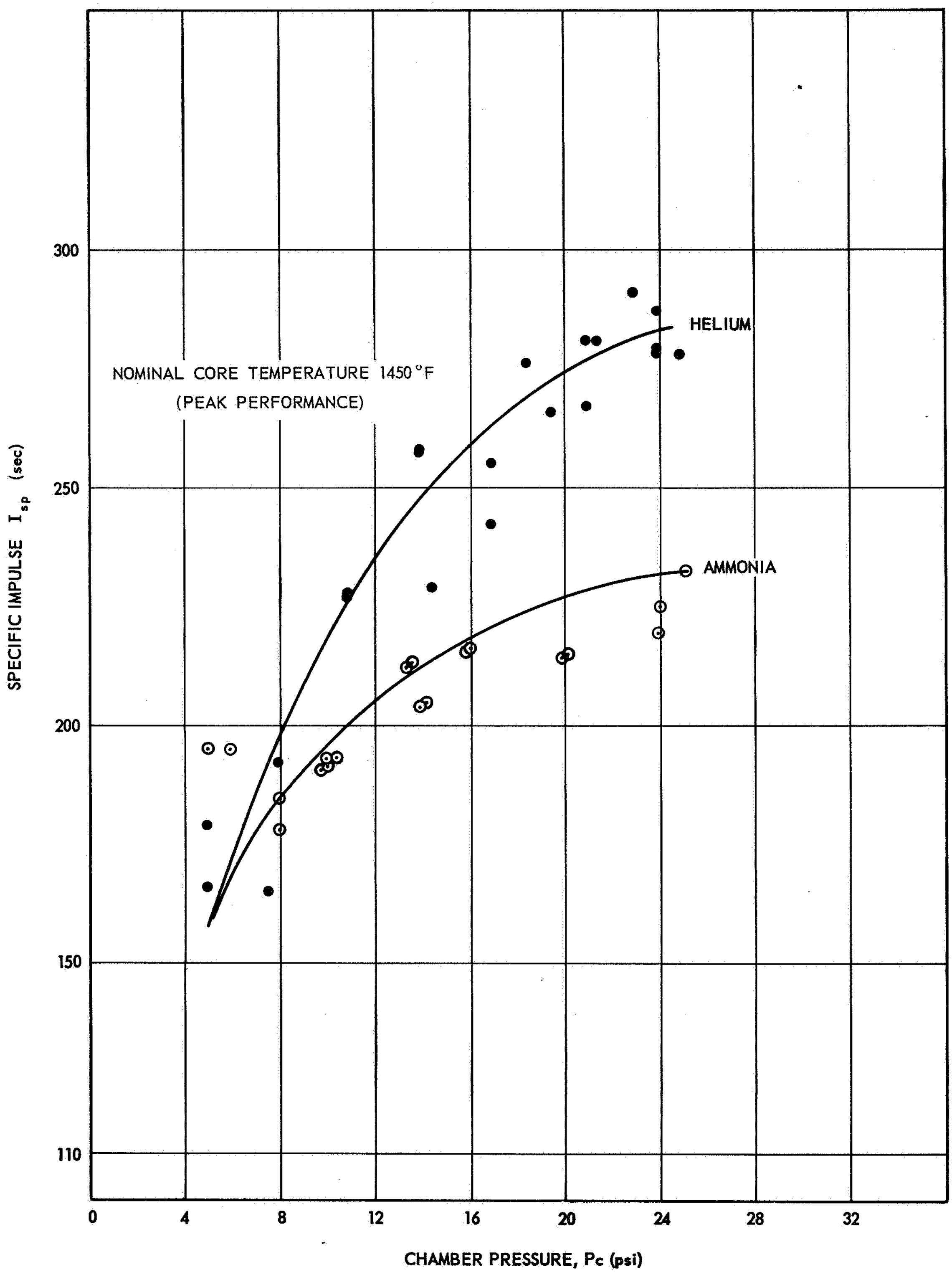


Figure V–8. Promethium-147 Fueled Thruster, Specific Impulse vs Chamber Pressure

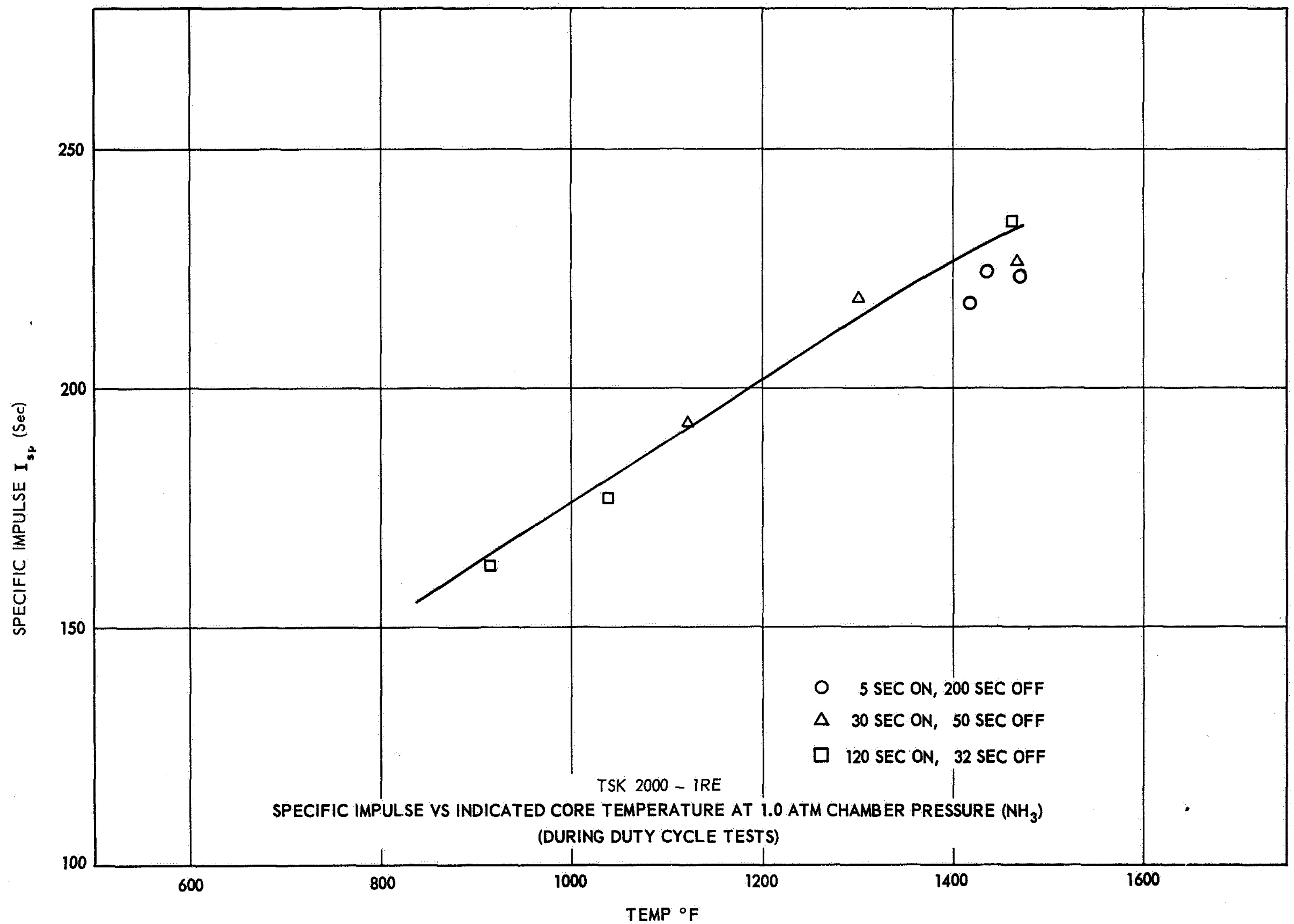


Figure V–9. Promethium-147 Fueled Thruster, Specific Impulse vs Indicated Core Temperature

questionable—indicating 10-25% higher thrust than predicted for the nominal 1-1/2 atmospheres operating pressure. The fueled data (Figure V-6) closely approximate a value of 20 millipounds of thrust predicted for 1-1/2 atmospheres chamber pressure. During these tests, the calculated maximum delivered specific impulse for the promethium fueled thruster was 232 seconds at a nominal core temperature of 1500°F.

Propellant gas temperature was not measured nor was the degree of gas decomposition obtained, thus it was not possible to make an accurate assessment of the thruster performance efficiency based on predicted results. The conclusion which can be drawn from the results of the fueled tests is that the thruster propellant temperature, at the nozzle inlet, was between 1450 and 1850°F, depending upon whether the degree of gas decomposition was closer to 100% or 50%.

Subsequent testing of the electrically heated thruster resulted in a specific impulse of 230 seconds at 1660°F core temperature (60 watts input) which, from the predicted results, indicates a degree of gas dissociation of approximately 70%. This value of dissociation is reasonable for the thruster configuration at a core temperature of 1660°F.

Comparing results on the basis of specific impulse, the 230 seconds obtained at a measured core temperature of 1600°F during the final electrical tests at GE corresponds almost exactly with the 232 seconds obtained at the 60 watts promethium-147 power input. The curve (Figure V-1) from the final Mound Laboratory tests correlates a core temperature of 1670°F with 60 watts. Thrust, mass flow rate, and specific impulse data obtained during the final GE tests at various temperatures and pressures are shown in Figures V-10 and V-11.

Another objective of the test was to obtain a correlation of temperature cool down profiles for various pulse modes with both the electrically heated thruster and the promethium-147 fueled thruster. In almost all cases during the electrical thruster tests, however, the resistance change of the heater wire varied (during long term "on" cycles) such that no constant power curve could be generated. In some cases, an increase in core temperature was noted after a series of pulses, primarily because the power input rose considerably while the thermal mass of the thruster body was being cooled.

With the promethium-147 thruster, on the other hand, the data obtained was consistent as shown in Figure V-12 and Appendix C. For each duty cycle, which is defined as (pulse-time-on)/(pulse-time-on plus pulse-time-off), a base steady state temperature was recorded as described in the S.O.P. These data points, plotted in Figure V-12, show a direct relationship between duty cycle (percent

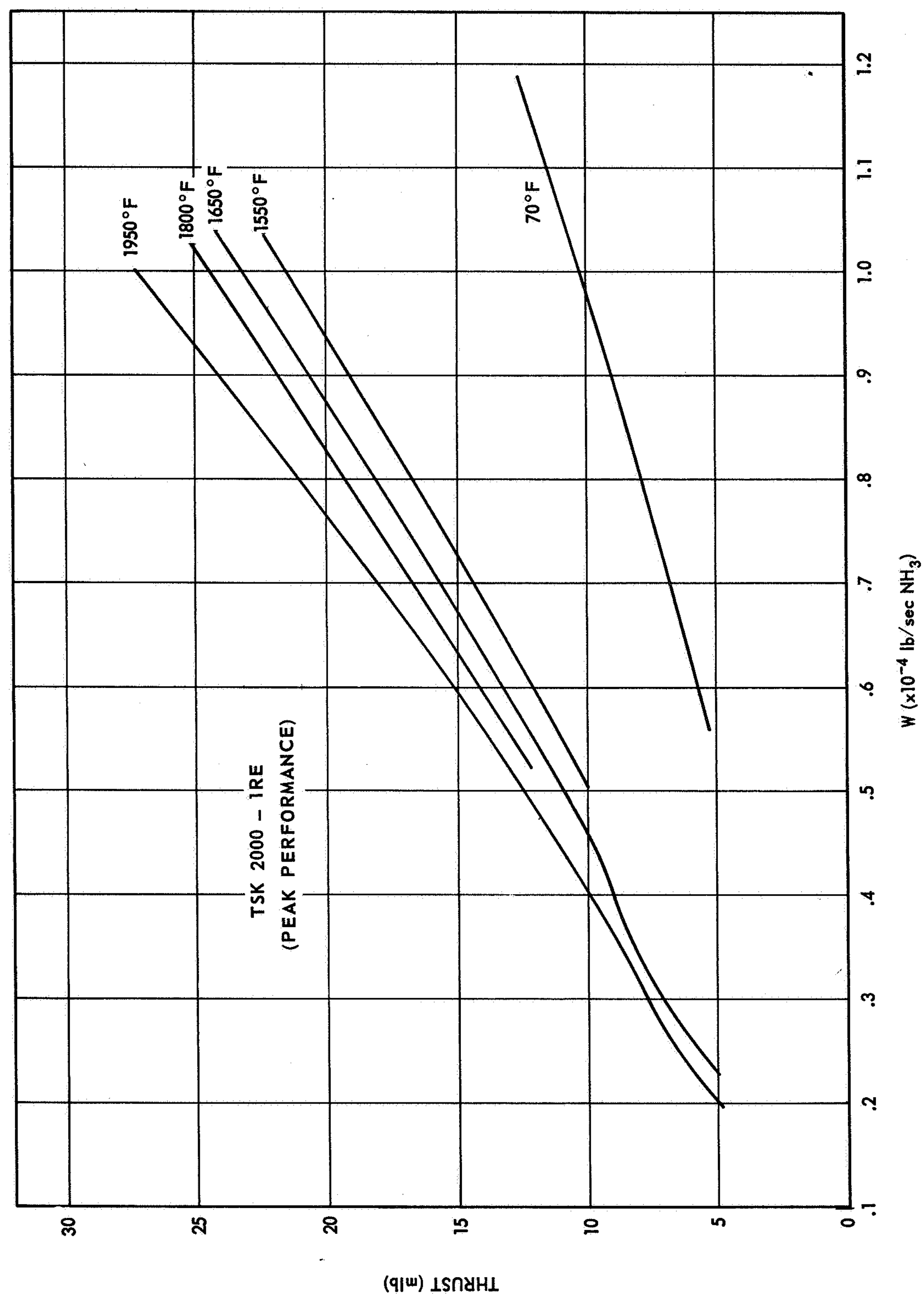


Figure V–10. Electrically Simulated Radioisojet, Thrust vs Ammonia Mass Flow Rate

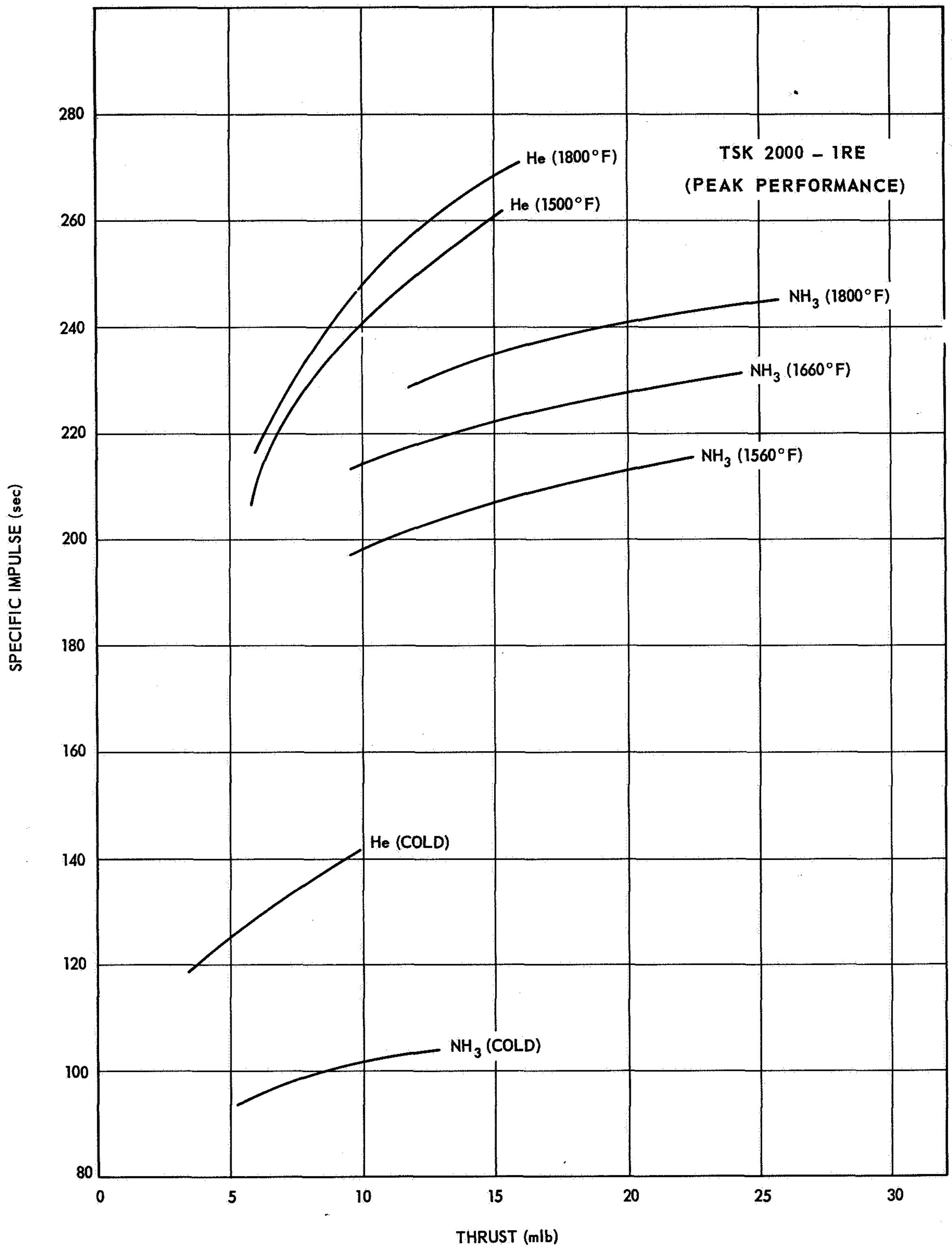


Figure V–11. Electrically Heated Testing at GE After Removal of Promethium-147 Capsule – Specific Impulse vs Thrust

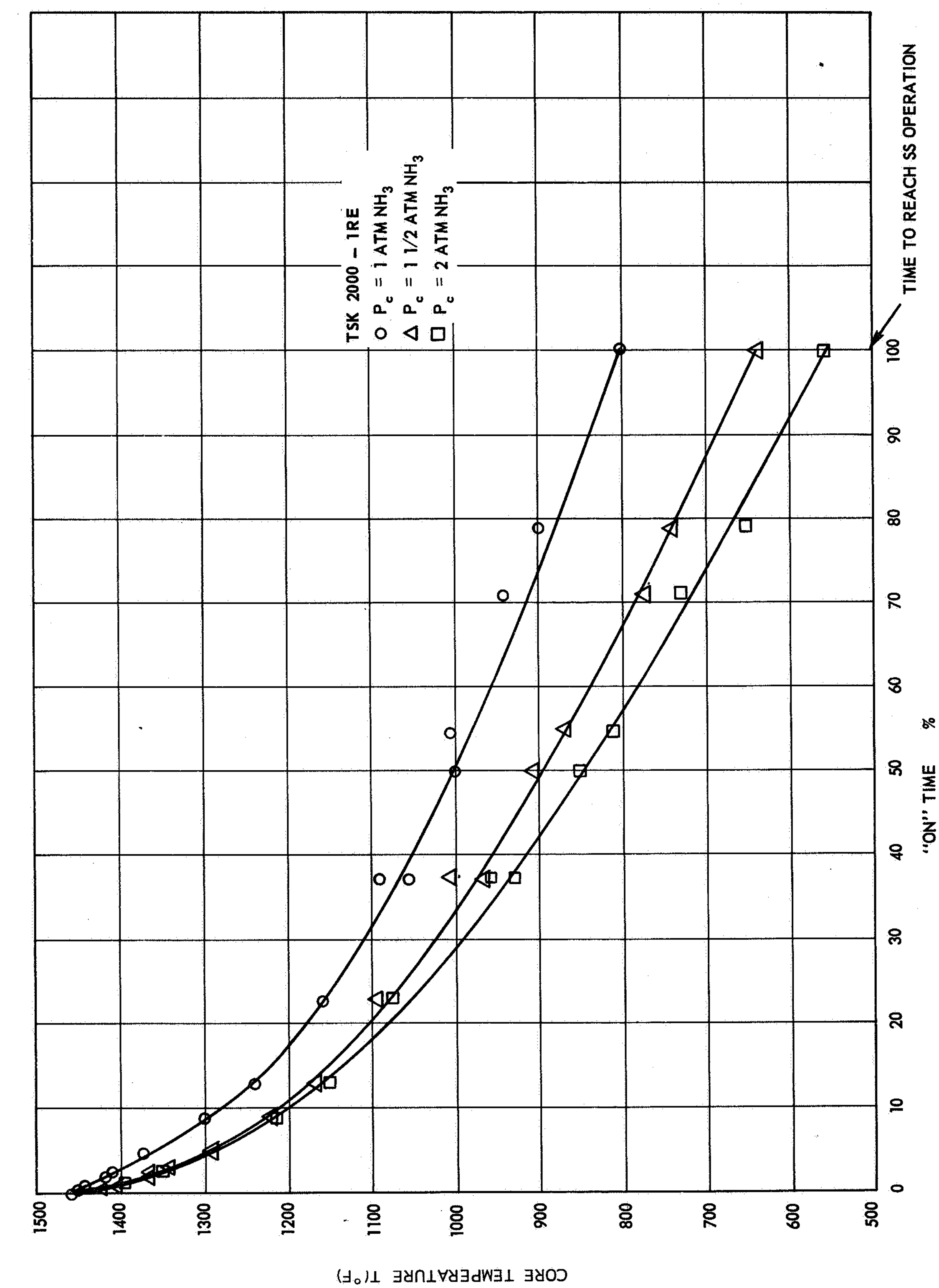


Figure V–12. Core Temperature vs Percent "On" Time

"on" time) and base temperature and chamber pressure. No effect of pulse duration is evident since the duty cycle points were plotted independently of pulse width.

C. POST TEST CAPSULE AND THRUSTER EVALUATION

Continued testing of the micro-thruster is being conducted at Battelle Northwest. The microthruster is tested under vacuum with a daily two-hour pulse of ammonia to produce maximum anticipated thermal shock as would be seen in actual operation. Testing will continue for approximately one year or until the cladding fails. It is proposed to remove the capsule at three or six month intervals for nondestructive examination and diagnosis to gain the maximum advantage from the test program. Nondestructive tests will include smear tests, dimensional measurements, helium leak tests, and possibly further chemical analysis of degraded surfaces. Companion ammonia-Hastelloy-X surface catalysis degradation tests during the program will be run as a monitor to capsule performance.

After 1/2 to 1 year of testing, it is proposed that the thruster will be given a second thrust test followed by destructive examination of the fuel capsule at Battelle Northwest. Essentially all the nondestructive tests performed on the capsule during fabrication will be repeated during dismantling to detect any changes and to correlate the data with capsule test conditions. Specifically, the post-test program will include:

(1) Layer by layer removal of the capsules by sectioning in nonradioactive facilities (same as one-third scale capsule analysis — cf. Appendix D). Smear tests, visual examination, dimension measurements, helium leak checks (if not contaminated), and radiography (if not contaminated) will be performed. The innermost capsule will be opened in contained facilities.

(2) Metallographic examination of:

(a) Stainless Steel to Hastelloy-X weld (capsule support to capsule)

(b) Hastelloy-X to Hastelloy-X weld (EB welded final capsule closure)

(c) Hastelloy-X to ammonia surface layer (along the axial length of the thruster)

(d) Hastelloy-X to inner capsule reaction zones (if any)

(e) Outer capsule to outer capsule weld (TIG plus EB closure of outer capsules)

(f) Inner capsule to inner capsule weld (TIG plus EB closure of innermost capsule)

(g) Inner clad — Pm_2O_3 compatibility

(h) Pm_2O_3 analysis (only if warranted by part "g" above).

(3) Chemical analysis of reaction zone products: semiquantitative spectrochemical analysis and/or vacuum fusion analysis depending on other tests.

VI. CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

At this point, a review of the RIJ Program objectives is warranted.

It must be emphasized that this was a R and D program to demonstrate the practicability of marrying a radioisotope fuel capsule with a low thrust level pulse jet thruster. The RIJ was not designed as or intended to be flight hardware. It was designed for a hot-firing ground test to demonstrate concept feasibility, performance capabilities, and to provide practical design, test, manufacturing and fabrication information. The primary objective of demonstrating concept feasibility was achieved. The promethium-147 fueled microthruster principle was verified and its performance capability demonstrated. In short, a technology ready date* has been attained. Subsequent to this achievement, however, a further development will be required to configure and qualify a system for a specific application.

Other objectives as outlined in the Introduction (Section II-B) are also considered to have been successfully accomplished. Test facilities and procedures have been defined, as have the required measurement parameters (to establish thruster performance). Specific RIJ design goals were attained, for the most part. However, maximum performance was not available, since the anticipated (desired) core temperature of 2000°F was not reached. The accomplishment of a successful test program without radiological incidents also demonstrates the safety associated with an isotopic system. The test program was performed in a timely manner with all scheduled milestones achieved. This was most gratifying to the program managers when considering the program involved several government agencies and numerous prime and sub-contractors.

B. RECOMMENDATIONS

The recommendations presented are based on the anticipated ultimate program goal of developing a flight qualified Radioisojet. In order to obtain this objective certain tasks must be accomplished. These tasks, which are listed below, were defined as a result of the present RIJ test program. In addition,

*Technology ready date is defined as that point in time when the major problems of a particular system are defined and/or solved, and, in some cases, a demonstration of the solution conducted.

this list is not to be construed as in order of precedence or chronology, but as a minimum effort to meet the anticipated program goal (i.e., flight qualified hardware).

(1) Define the mission, i.e., station keeping, orbital transfer, and/or attitude control.

(2) Conduct a thorough literature search on such topics as: gas decomposition, catalytic surfaces, heat shields (construction, performance, etc.).

(3) Establish a definitive analytical background tc include, at least, the following:

(a) The primary dependent and independent variables necessary to define the system analytical model,

(b) A parametric study to establish optimum (max. I_{sp}, F_o, etc.) design criteria for specified missions,

(c) An aerospace nuclear safety study,

(d) The thruster theoretical thermal profile,

(e) A preferred heat shield design based on the literature search,

(f) Determine the "real" (or expected) system thermal profile (based on empirical data obtained during component testing); i.e. develop an empirical analytical model.

(4) Supplement the present RIJ systems tests with the following:

(a) Component tests:

(1) Perform a test program to determine the optimum catalytic ingredients to obtain the highest degree of gas decomposition over the desired operating temperature range by varying surface material, temperature, flow rate, and propellants.

(2) Verify the selected heat shield design through performance tests.

(3) Verify optimum thruster performance, i.e., F_o and I_{sp} for various temperatures, flow rates (pressures), power input, etc.

(b) Perform additional systems tests (as necessary) to verify the empirical model developed as a result of the above specified tasks.

The above recommendations are suggested in order to enhance the present RIJ program and bring this now proven concept to the point of operational acceptability. The basic premise, of course, is the successful continuation of tests now being conducted at Battelle Northwest and a satisfactory destructive examination of the RIJ fueled capsule following the proposed second test effort.

VII. REFERENCES

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4. W. V. DeMier, G. L. Richardson, A. M. Platt, and W. H. Swift, Development of Radioactive Waste Fractionization and Packaging Technology, HW-SA-2786, General Electric Company, Richland, January, 1963.

5. L. R. Bunney and E. M. Scadden, Decay Characteristics of Pm-146, USNRDL-TR-1109, November 1966.

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APPENDIX A

THE TSK 2000-1RE RADIOISOJET

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APPENDIX A

THE TSK 2000-1RE RADIOISOJET

1. Engineering Drawing, TSK 2000-1RE Radioisojet (Figure A-1)

2. Radioisojet Fabrication-Welding

The Pm_2O_3 encapsulation development program for this specific application and concept study resulted in a process (Figure A-2) which included:

(1) Promethium purification,

(2) Sintering pellets of Pm_2O_3 for fuel core densification,

(3) Double encapsulation of the fuel using TIG and EB welding techniques,

(4) Assembly of the encapsulated heat source into the inner and outer thruster bodies.

Each step in the process was appropriately characterized (Figure A-2) including:

(1) Isotopic and chemical purity analyses of the fuel,

(2) Dye penetrant, ultrasonic integrity, helium leak detection, radiography and metallography of the cladding materials and the various welds,

(3) Dimensional control checks,

(4) Decontamination smear tests,

(5) Radiation mapping, gamma spectrometry and calorimetric measurements.

The program also included 45 day tests of two one-third scale Pm_2O_3 capsules at 1100°C and examination of the thruster after thrust tests at Mound Laboratories.

a. Welding — Two fabrication welds, each involving Hastelloy X to 304 stainless steel, were made in assembling components of the microthruster. The shell was welded to the shell support and the capsule shroud was welded to the

capsule support (see Figure A-3). Both were accomplished by manual TIG welding in the inert gas weld chamber. In each case, several sets of simulated parts were fabricated and used to arrive at satisfactory parameters, as determined by sectioning and metallographic examination.

The inner and outer liners, fabricated from powder metallurgy product, were rolled and seam-welded to form the capsules. The seam welds on the cans, were made by a semi-automatic gas tungsten-arc process. However, nondestructive testing of the capsule components showed porosity in the welds. A manual TIG (tungsten inert gas) overpass was made on a sample seam weld to reduce porosity, but comparative testing showed an increase in weld porosity. Consequently, the capsule containing a minimum weld porosity was selected for use with the promethium fuel. Metallography of this capsule showed the parent metal to be in a well annealed condition.

Both TIG and electron beam (EB) welding techniques were used in the fabrication and sealing of capsule components. Peripheral TIG welds were made on the capsule end closures (Figure A-4). The EB welder was used to drill outgas holes in the final endcap prior to assembly and to close these outgas holes after the TIG peripheral weld (Figure A-4).

Capsule components were originally supplied with bottom end caps of a 0.060 inch thickness welded in place. These were replaced with end caps of 0.030 inch thickness to reduce the overall length of the finished capsule. The endcap welds were made by a manual TIG process in the inert gas chamber. A vertical rotary weld fixture was used to rotate the pieces within the chamber while the weld current, torch position, and rotational speed were varied manually. A typical cross section shows minor porosity on one side (Figure A-5). The major amount of porosity, but not all, appeared in the endcap portion of the heat affected zone. Some larger pores and also porefree areas in these welds were observed (Figure A-6a and A-6b). Radiography showed that all the welds had varying amounts of porosity. The final welded outer capsule is shown in Figure A-7a.

The peripheral weld on the second endcap (after fuel loading) was performed with the capsule held in a heavy copper chill block. The chill block was placed at one-sixteenth inch from the top of the can.

Outgas holes in capsule endcaps were formed by drilling with the electron beam. A continuous beam of approximately 110 KV and 6 to 7 ma current for two to three seconds was required to form a hole in 0.030 inch thick material. This technique invariably caused cratering on both sides of the cap. By using a pulse technique (above parameters approximately the same) at a frequency of

45 cps and a pulse width of 5 msec the cratering was eliminated. Holes varying in diameter from 0.005 inch to 0.020 inch were made. Closing of the outgas hole was accomplished by manually controlling the magnitude and sweep of a deflected beam (approximately 110 KW and 5 ma beam current) Figure A-7b.

b. Heat Source Assembly — The doubly clad Pm_2O_3 heat source was inserted into the Hastelloy X (Hx) shroud. The final seal weld on the Hx shroud was made by an EB semi-automatic welding process. Again, simulated test pieces were used to set parameters. A 2 ma beam of 115 KV potential was used to make the single pass Hx to Hx joint at a weld speed of 30 ipm.

Thruster assembly data consisted of radioactive contamination smear checks (where appropriate) dye penetrant tests, radiography, metallography, radiation mapping and dimension measurements (Table A-1). Both stress and shrinkage cracking were observed in the Hx to Hx dome to thruster body test weld (Figure A-8).

The specified welding process for the shell to shell support weld should be changed to EB since TIG welding produced severe distorting in this joint design as indicated in Table A-1.

The final capsule and the thruster body assemblies are shown in Figures A-10 and A-11, respectively. Component weights were as follows:

Capsule assembly with back heat shield and thermocouple		856 grams
Outer Thruster Body		465 grams
Heat Shield Assembly		1127 grams
	Total	2448 grams

3. Promethium Purification

The rare earth elements resulting from nuclear fission and decay are isolated from plant waste by utilizing a series of successive precipitation processes (References A-1 and A-2). After lag storage to allow for decay of the gamma emitting Pm-148, the rare earth crude is further refined by solvent extraction (Reference A-2), thermal concentration and "acid kill" (Reference A-3) steps prior to transfer to the Hot-Cell Facility.

Table A-1

Thruster Assembly Test Results

	Thruster Shell, Shell Support, Capsule Support (Figure A-9)	Hx Top Dome	Hx-S/S Test Weld	Hx Top Dome to S/S Capsule Support	Hx Inner Thruster Body	Hx-Hx Test Weld	Hx Top Dome to Hx Inner Thruster Body[d] (Figure A-10)	Hx-S/S Test Weld	Hx Thruster Shell to S/S Shell Support (Figure A-9)	Final Assembly
Radioactive Contamination Smear Tests	N.A.[a]	N.A.	N.A.	N.A.	N.A.	N.A.	non-smearable	N.A.	N.A.	non-smearable
Dye Penetrant Tests	no defects observed[b]	no defects observed	no defects observed	no defects observed	no defects observed	no defects observed	no defects observed	no defects observed	no defects observed	no defects observed
Radiography	no defects observed	no defects observed	no defects observed	no defects observed	no defects observed	no defects observed	—	—	—	—
Metallography	N.A.	N.A.	full weld penetration (Fig. 15)	N.A.	N.A.	some cracks observed at weld root in thick section (Fig. 16)	N.A.	full penetration weld	N.A.	N.A.
Radiation Mapping	N.A.	N.A.	N.A.	N.A.	N.A.		48 mr/hr at 0.5 m from side of capsule	N.A.	N.A.	See Fig. III-10
Dimensions	[c]	1.188" OD 0.913" length TC hole bottom 0.115" from flat surface (inside capsule)	N.A.	[c]	1.188" OD 0.025" wall 2.708" inner length		1.093" OD at weld	—	0.010" TIR[e] on flange 0.017" TIR in shell throat	—

(a) N.A. = not applicable

(b) capsule support not tested

(c) per GE drawing 246 R 711 Rev. C (verified)

(d) Pm_2O_3 heat source capsule in place

(e) Total Indicated Runout

Isolation of promethium from its adjacent rare earth neighbors and all other impurities is accomplished by a displacement cation exchange process (Reference A-4) utilizing an ammonia buffered solution of the complexing agent DTPA* as the separating agent. The separation is conducted in a 7-column ion-exchange facility, installed, together with associated tanks, valves and instrumentation, in a heavily shielded manipulator cell. The flowsheet and column arrangement is shown schematically in Figure A-12. The promethium used for this capsule was taken predominately from two separation runs completed on September 5, 1966 and September 14, 1966.

With the very high Ce-144 and Eu-154 decontamination factors achieved in the ion-exchange process, pure promethium does not require extensive shielding for the final conversion to treated oxide. The promethium product solution is therefore collected as it flows from the final purification column and transferred to the lightly shielded glove box facilities for oxide conversion.

The oxide conversion chemistry process consists of the following operations:

(1) Vacuum transfer of up to 9 liters of promethium produce solution (~ 45 gms Pm) from the purification facility to a glass precipitator in the glove box,

(2) Heating of the promethium solution to 80°C via a steam coil in the precipitator and simultaneous sparging,

(3) Vacuum addition of sufficient oxalic acid (as a slurry) to ensure complete precipitation of the promethium as oxalate,

(4) A 30 minute digestion period followed by cooling to 25°C (cooling water through the coil) to promote growth of large crystals and lower solubility,

(5) A double wash by the decantation of the precipitate to remove traces of stainless steel corrosion products acquired during storage in the steel product storage tanks,

(6) Slurry transfer of the precipitate to a quartz filter tube containing a quartz frit at the bottom,

*Diethylenetriaminepentaacetic acid.

(7) Insertion of the loaded filter tube into a vertically mounted tube furnace followed by an 8-hour heating cycle at 1100°C to convert the oxalate precipitate to Pm_2O_3,

(8) A special oxide treatment step consisting of 4 hours at 1100°C with oxygen flowing up through the filter tube and contents at a slow rate, followed by a similar treatment with 6% hydrogen-94% argon to ensure removal of final traces of bound CO_2 and yield a Pm_2O_3 product slightly deficient in oxygen.

Radiochemical analysis, corrected to October 28, 1966, indicated the Pm_2O_3 capsule contained:

Pm-146 — 3.6×10^{-2} curies (Pm-146/Pm-147 = 2.2×10^{-7} curies/curies)

Pm-147 — 164,100 curies

Pm-148m — 1×10^{-4} curies (Pm-148m/Pm-147 = 7×10^{-9} curies/curies)

Eu-154 — 4.1×10^{-3} curies (Eu-154/Pm-147 = 2.5×10^{-8} curies/curies)

The samarium-147 daughter consistuted the principal impurity, approximately 3%, on the above date. Non-rate earth impurities spectrographically detected were Al, Pb, Y and Si and constituted less than 1% of the oxide. X-ray analysis of the treated oxide confirmed the crystal structure to be 100% monoclinic.

APPENDIX A

REFERENCES

1. L. A. Bray and H. H. Van Tuyl, Laboratory Development of a Carrier-Precipitation Process for the Recovery of Strontium from Purex Wastes, HW-69534, General Electric Company, Richland, Washington, May 1961.

2. W. V. DeMier, G. L. Richardson, A. M. Platt, and W. H. Swift, Development of Radioactive Waste Fractionization and Packaging Technology, HW-SA-2786, General Electric Company, Richland, Washington, April 1963.

3. L. A. Bray, Denitration of Purex Wastes with Sugar, HW-76973 Rev., General Electric Company, Richland, Washington, April 1963.

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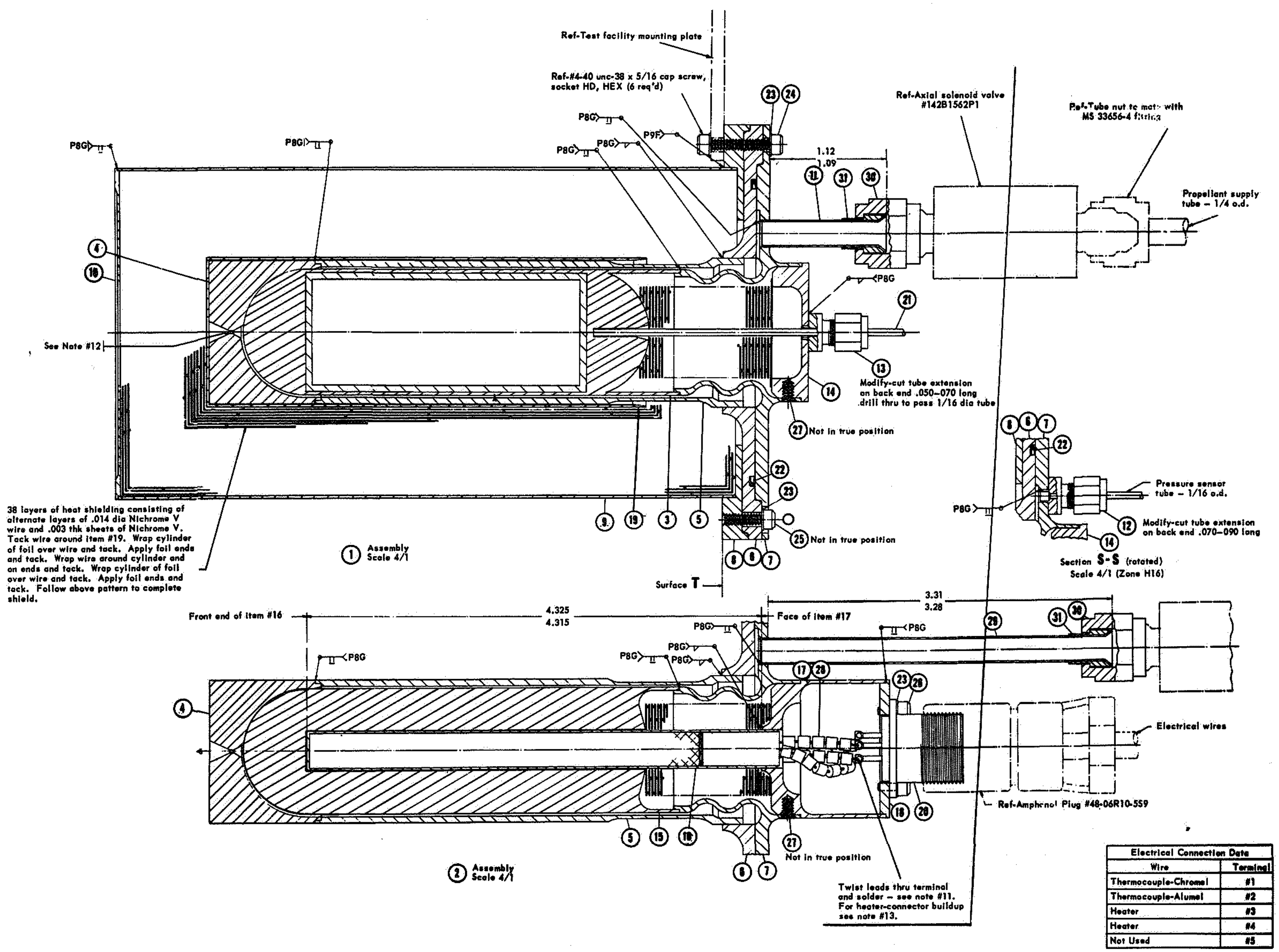


Electrical Connection Data	
Wire	Terminal
Thermocouple-Chromel	#1
Thermocouple-Alumel	#2
Heater	#3
Heater	#4
Not Used	#5

Figure A–1. Engineering Drawing, TSK 2000-1RE Radioisojet

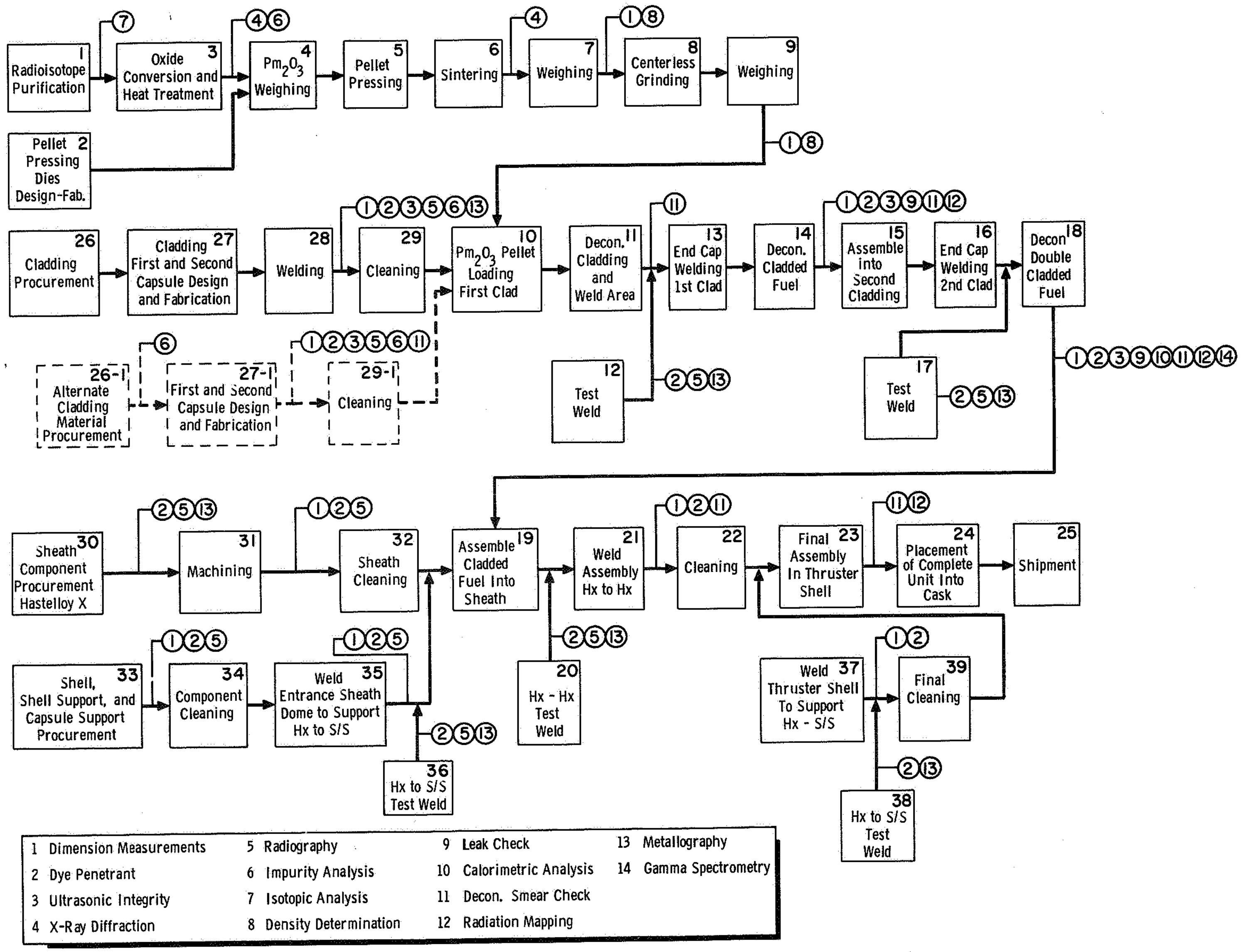


Figure A-2. Radioisotope-Fueled Microthruster – Capsule Fabrication Flow Diagram

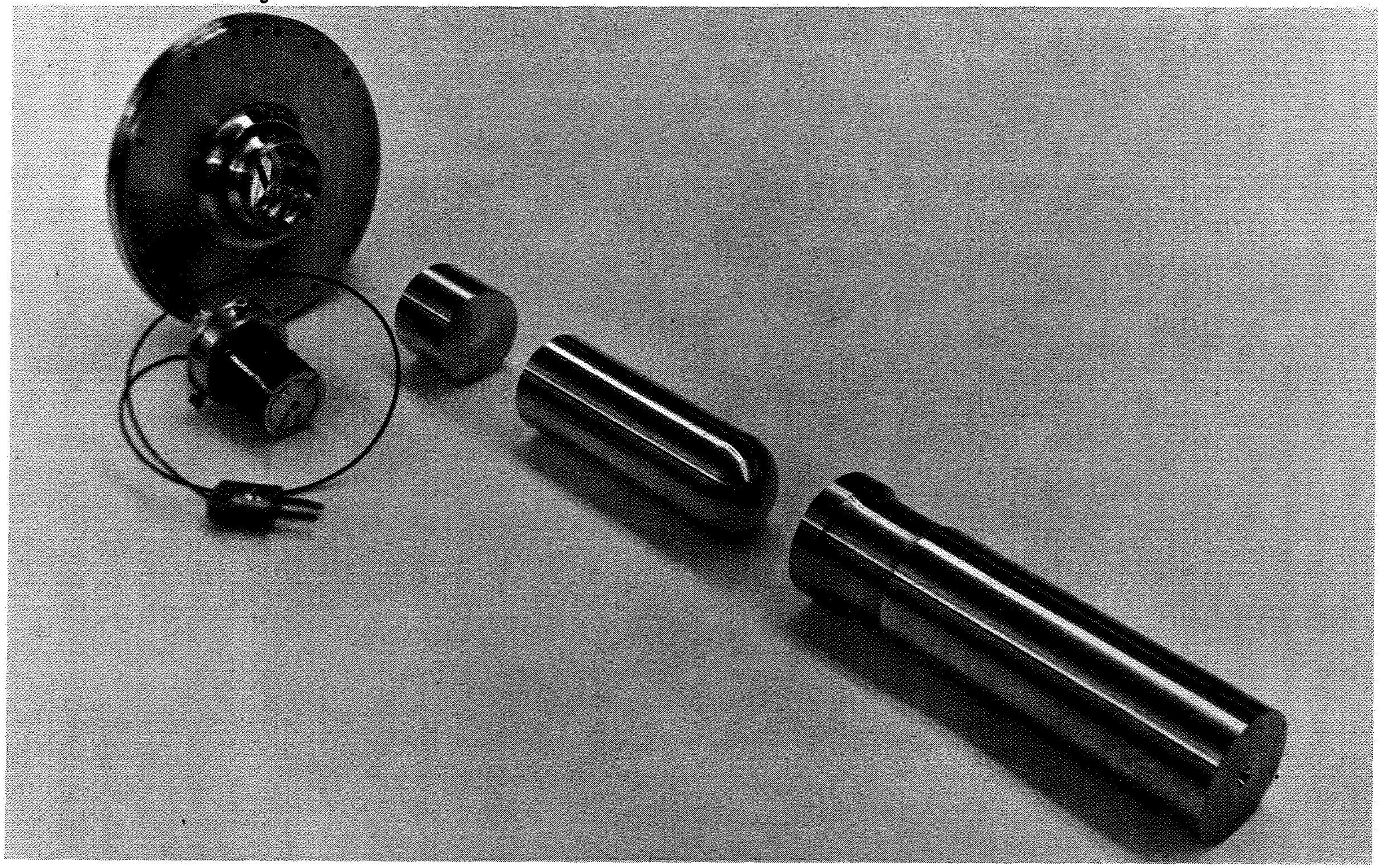


Figure A–3. Thruster Assembly

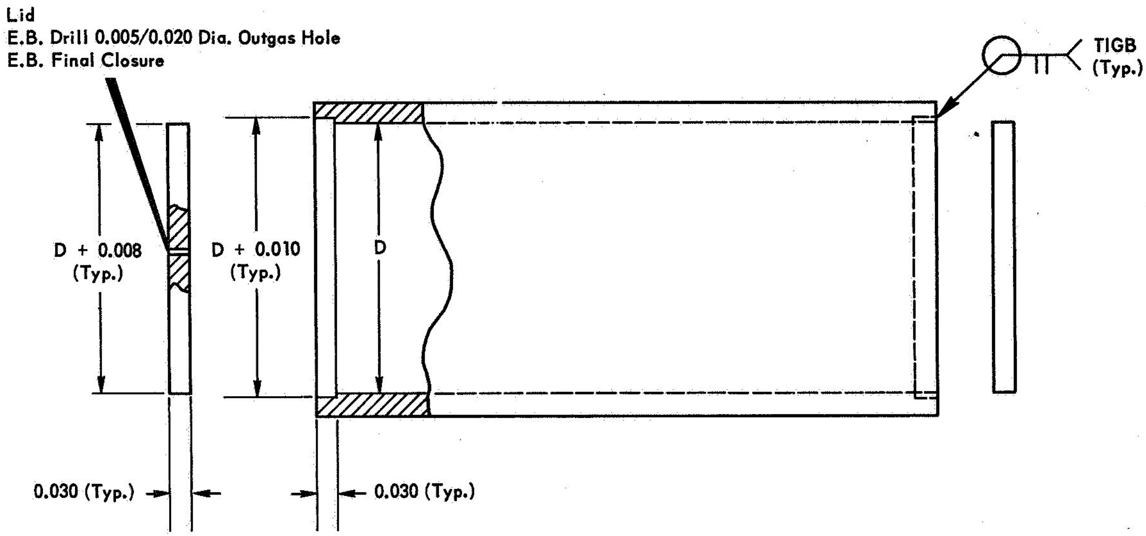
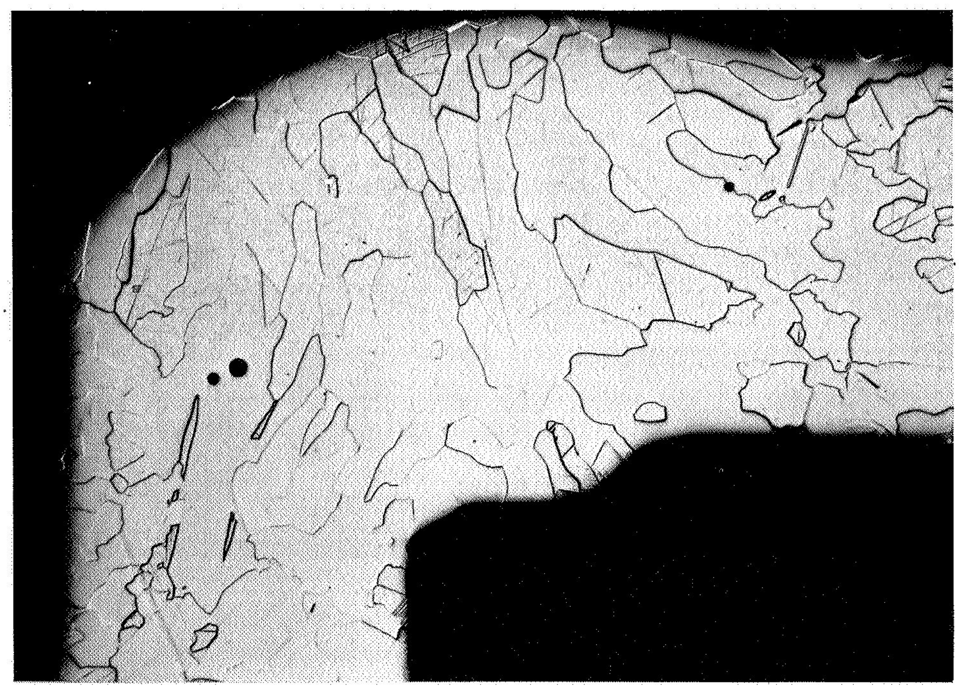


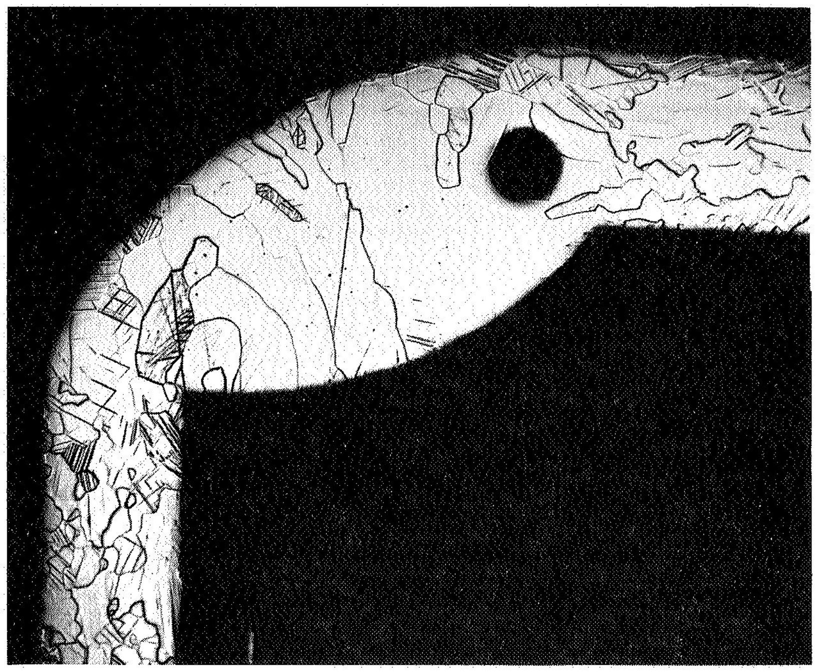
Figure A-4. Weld Design for Full Scale Microthruster Capsules



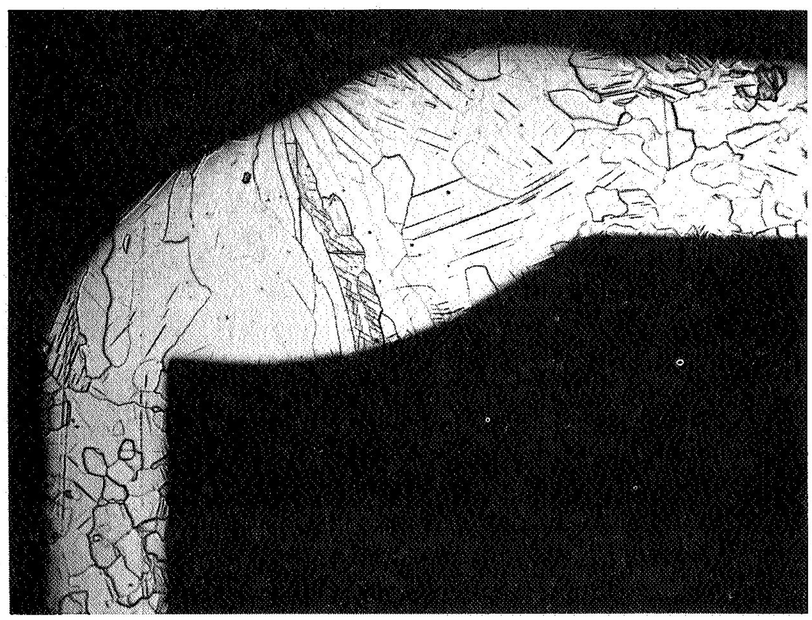
Magnification: 75X

Etchant: Murakamis

Figure A-5. Capsule Liner Test Weld



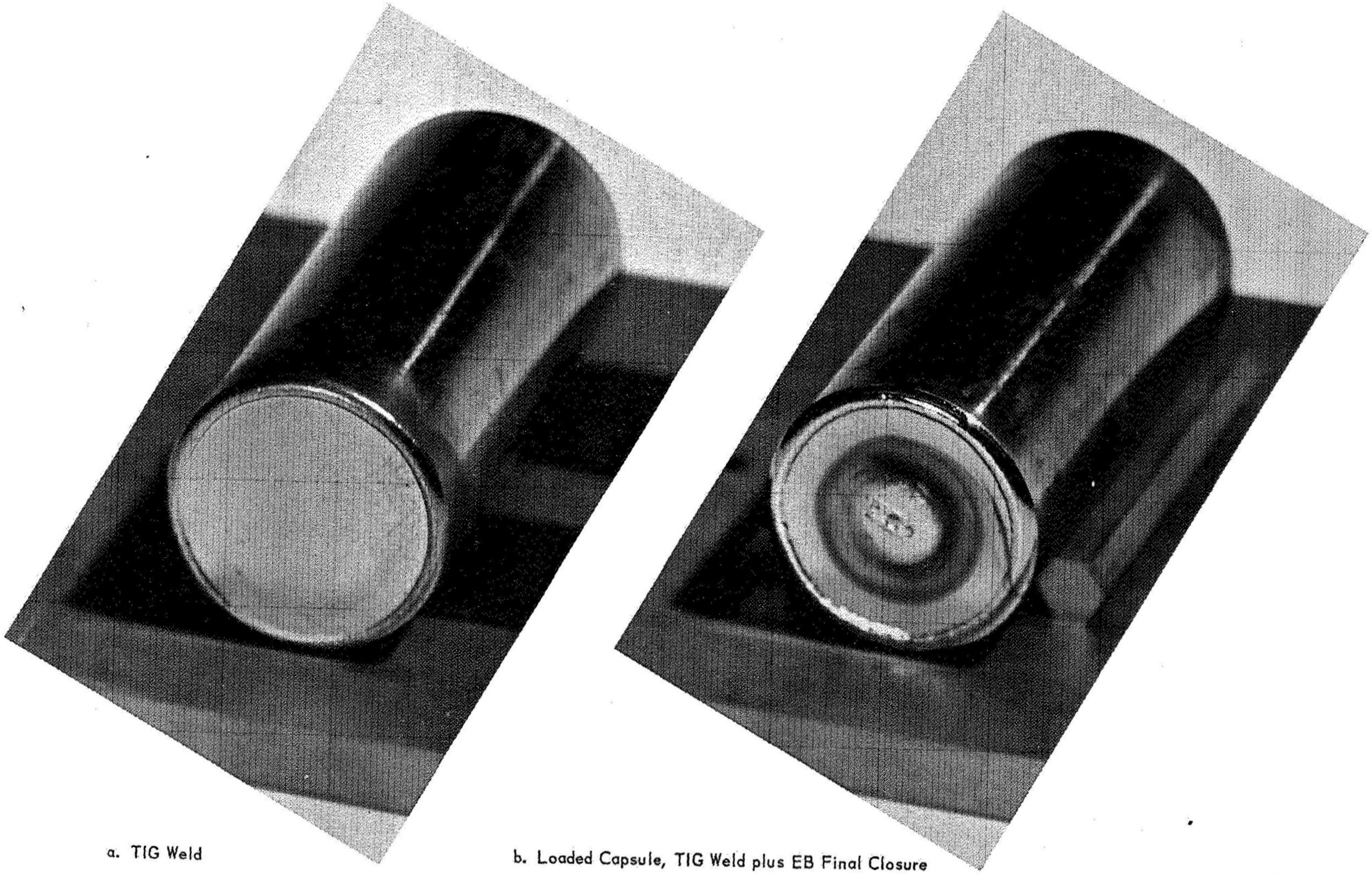
a. Weld Porosity



b. No Weld Porosity

Magnification: 36X Etchant: Murakamis

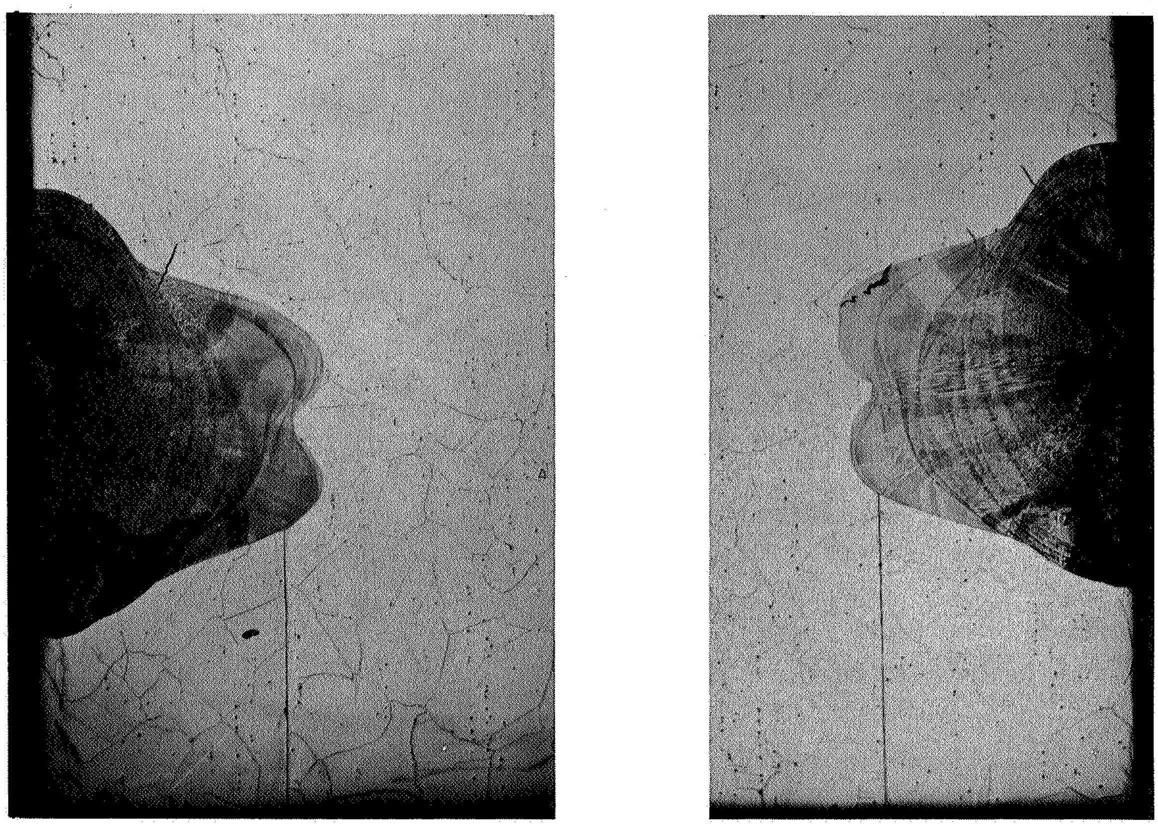
Figure A–6. Capsule Liner Test Weld



a. TIG Weld

b. Loaded Capsule, TIG Weld plus EB Final Closure

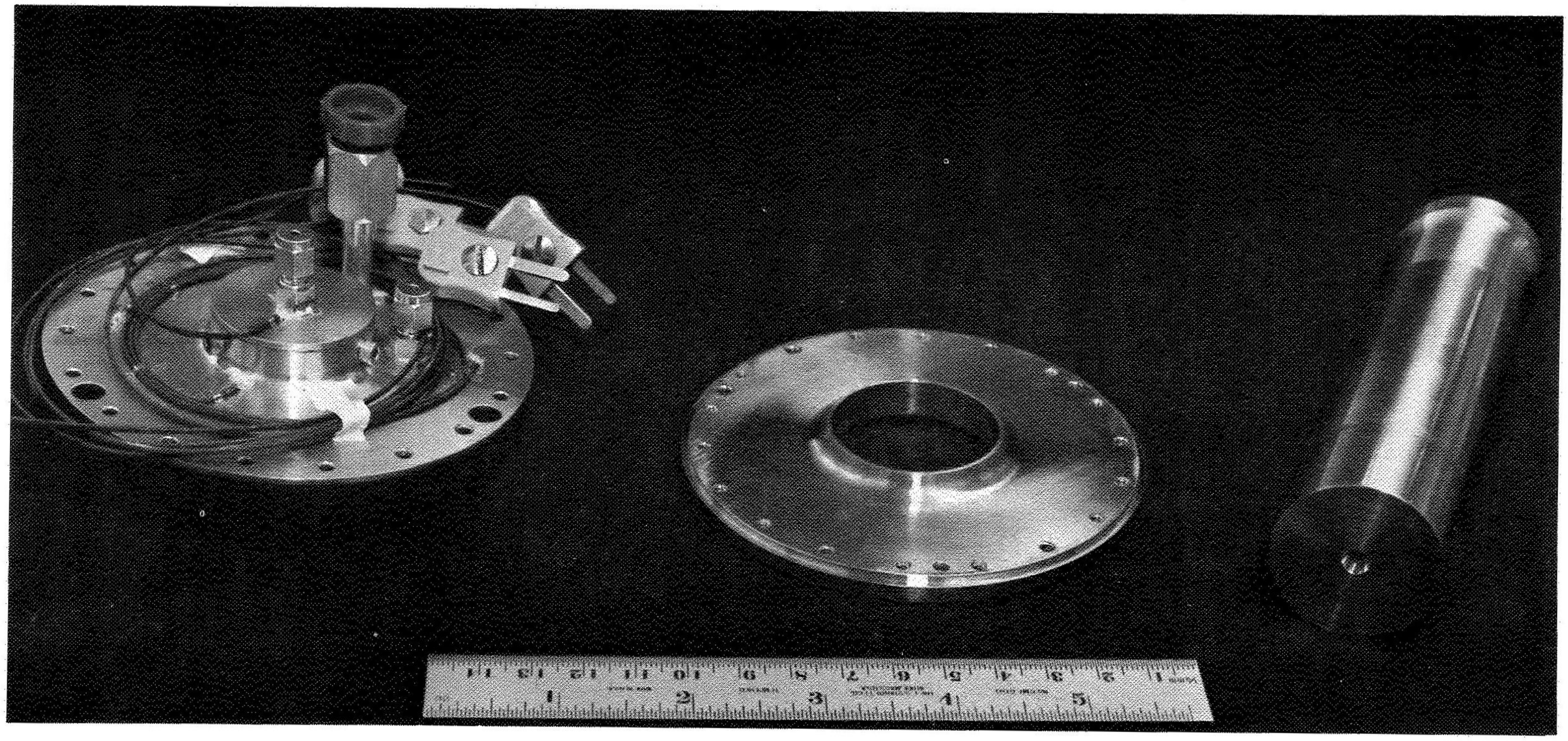
Figure A–7. Doubly Clad Pm_2O_3 Heat Source Capsule



Magnification: 36x

Etchant: HCL + H_2O_2

Figure A-8. Hx Top Dome to Hx Inner Thruster Body Test Weld



Inner Thruster Body
Support Flange

Outer Thruster Body
Support Flange

Outer Thruster Body

Figure A–9. Thruster Support Flange and Outer Thruster Body

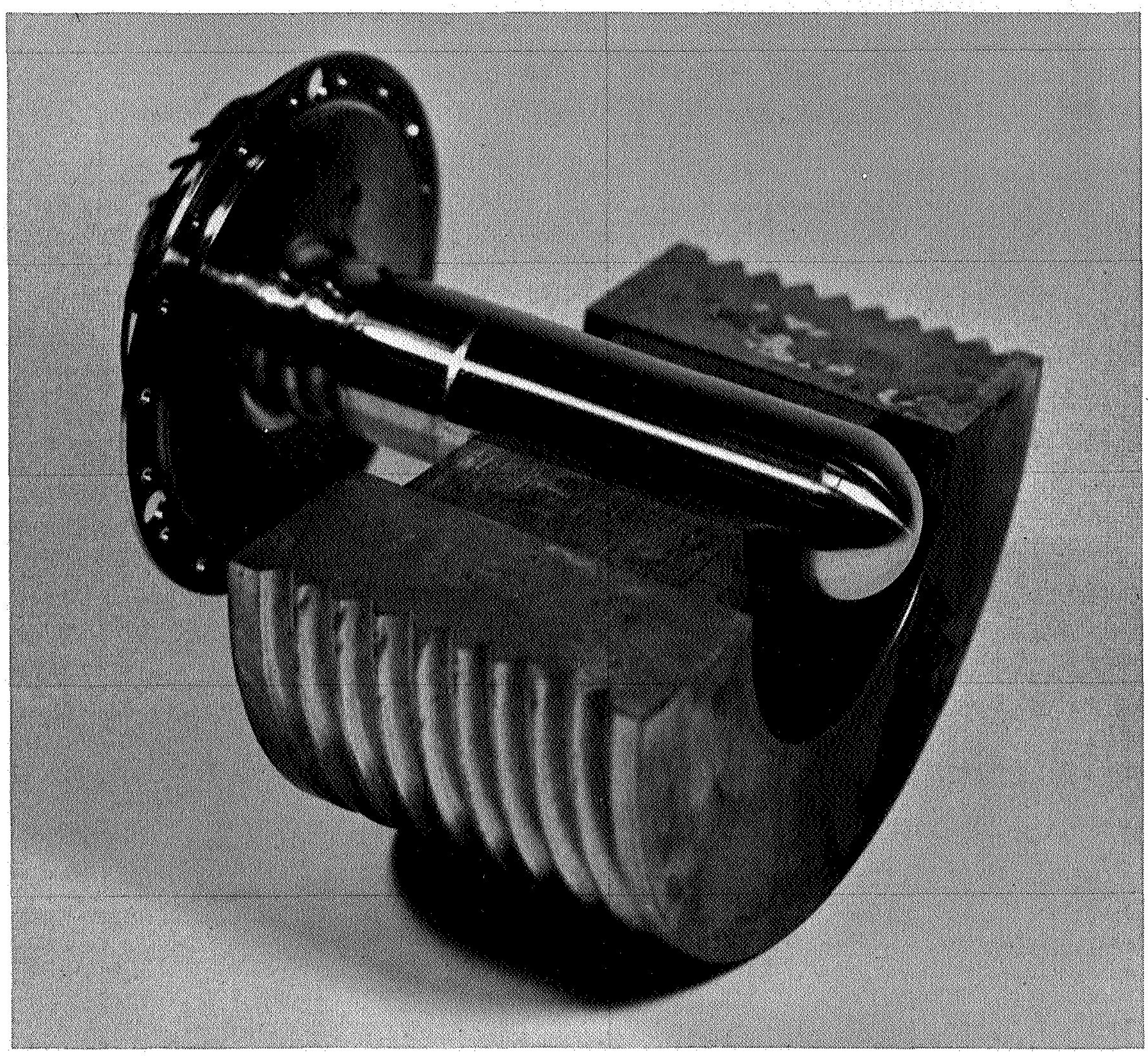


Figure A–10. Completed Microthruster Heat Source Assembly

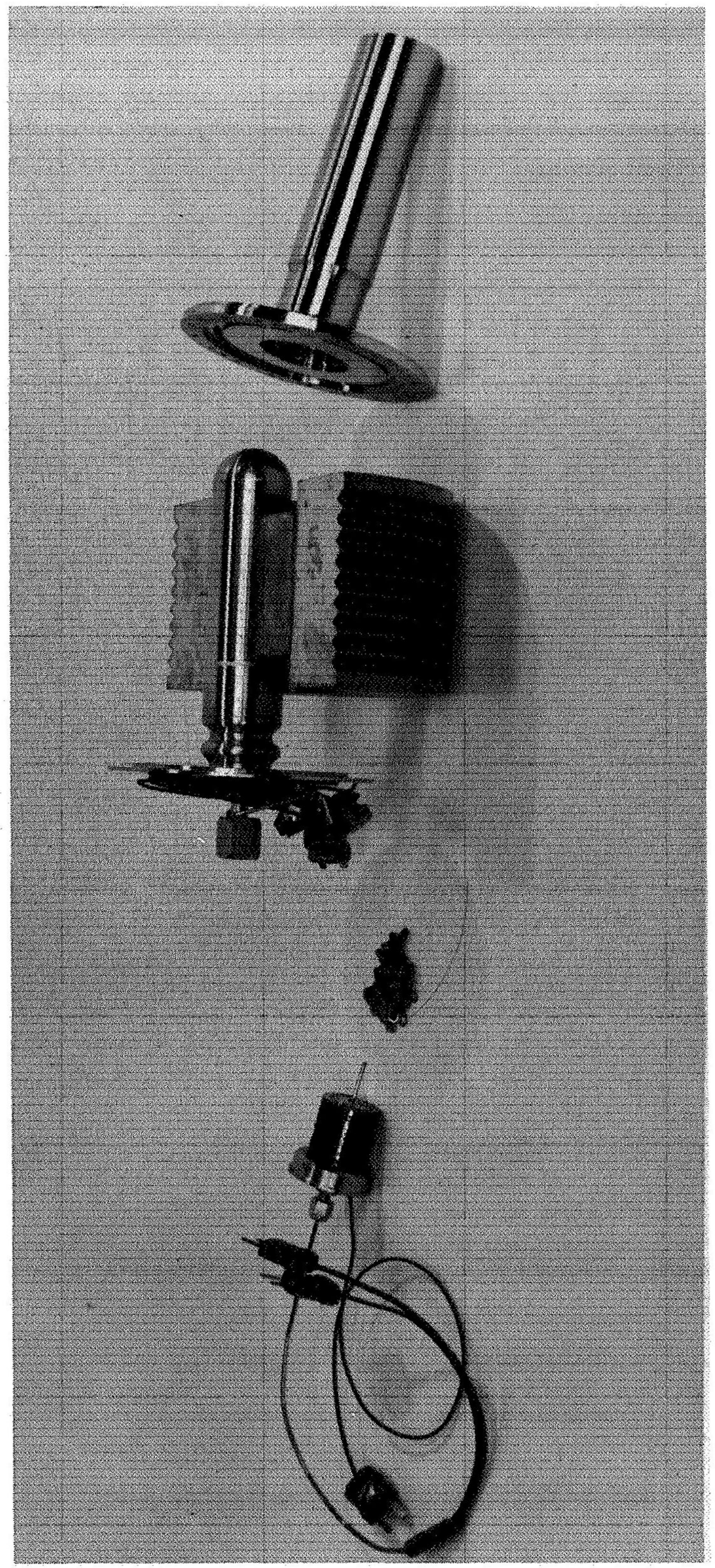
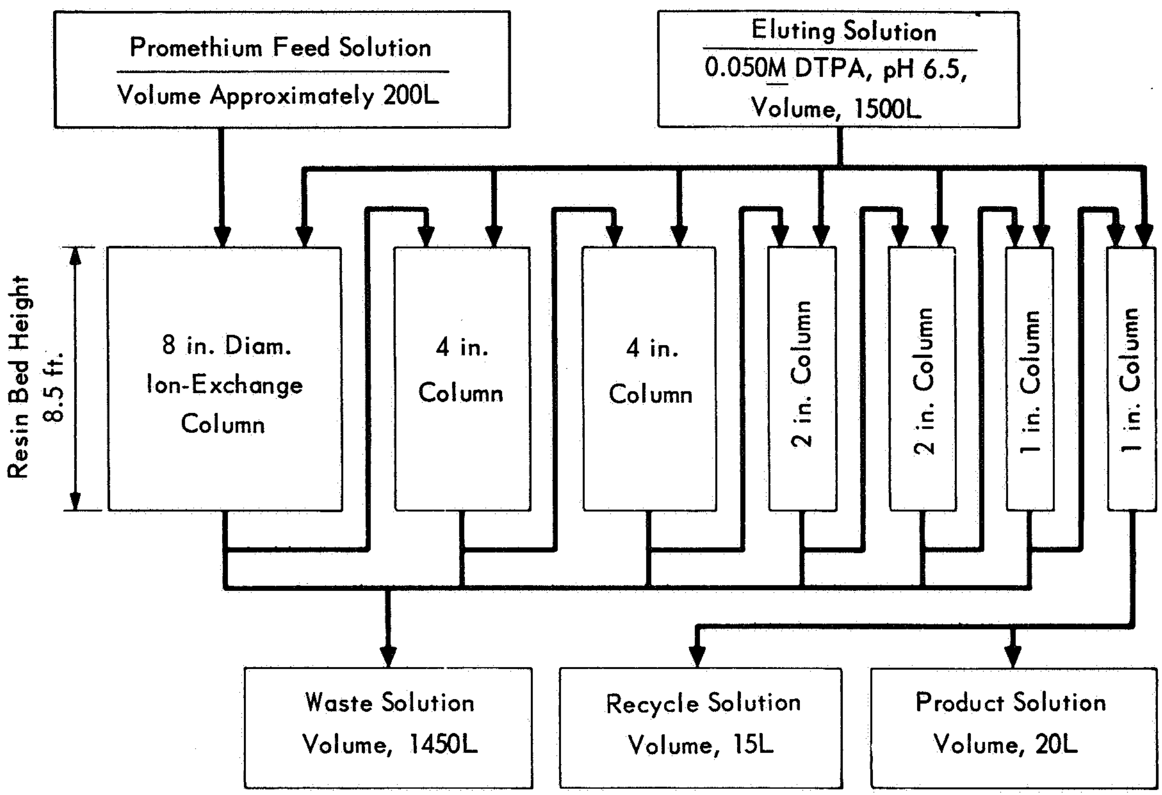


Figure A-11. Microthruster Assembly



Run Conditions:

Elution Temperature	65 C
Elution Flow Rate	4 ml/min-cm^2
Rate of Band Advance	16-18 cm/hr

Run Cycle Time

Absorption Cycle	4 hr
Elution Cycle	160 hr
Turn Around Time	72 hr

Figure A-12. Promethium Purification Flowsheet

APPENDIX B

STANDARD OPERATING PROCEDURES

Radioisotope Fueled Microthruster

APPENDIX B

STANDARD OPERATING PROCEDURES

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APPENDIX B

STANDARD OPERATING PROCEDURES

I. FACILITY AND INSTRUMENTATION REQUIREMENTS

The following test plan presents an outline of the general test information required to evaluate the performance of both electrically simulated and radioisotope fueled thrusters. This outline also ensures conformance to a Standard Operating Procedure (SOP) if repetitive tests occur subsequent to this report. Test instrumentation includes power supplies, signal generator, sensors and appropriate readout equipment for measurement (and operating ranges) of:

(1)	Propellant supply	(6×10^{-5} lbs/sec to 2×10^{-4} lb/sec)
(2)	Thruster chamber pressure	(0–150 psia)
(3)	Propellant and thruster body temperatures	(0–2000°F)
(4)	Vacuum system pressures	(ultimate to 1 atmosphere)
(5)	Thrust	(0–0.10 lbs)
(6)	Electric current	(0–4 amps)
(7)	Voltages	(0–30 volts)

The vacuum system must be capable of maintaining a vacuum of 4×10^{-4} torr or less during thruster operation with an ammonia flow rate of 1×10^{-4} lbs/sec.

The required electronic equipment is:

(1) Veeco Vacuum Gauges—Type RG-2A and RG-3A

(2) Veeco Thermocouple Gauge—TG-6 (2 Units)

(3) RIDL Radiation Monitoring Equipment (3 Units)

(4) Sanborn Strip Chart Recorder—Model (350)

(5) Tektronix Type 161 and 162 Wave Form and Pulse Generator respectively.

(6) Leeds and Northrup Speed-o-Max Model W, 12-point Strip Chart Recorders (4 Units).

(7) Daytronics Displacement Transducer and Amplifier—Indicator, Type 60—Tran. Amf. Ind. with Differential Transformer Plugin Unit—Model 300C.

(8) Pace—Pressure Transducer with Carrier Demodulator.

Flow charts of process, water, and electricity are included as well as a diagram of the test control panel.

II. PRELIMINARY ELECTRICAL TEST

A. Thruster Installation

1. Thruster Inspection.

   a. Dismantle thruster into its major components.

      i. The heat shield package

      ii. The thruster body

      iii. The heat source capsule and support flange

   b. Perform visual inspection in clean area.

      i. Examine Apex seal for indication of damage or non-uniformity.

      ii. Clean, if required, seal and seal mating area with acetone, removing any foreign particles before reassembly.

      iii. Check both heater element and thermocouple for continuity. Record resistance of each, at room temperature.

      iv. Check continuity of valve coil. Examine leads for damage.

   c. Reassemble thruster per GE Drawing 246R711.

2. Installation

   a. Install thruster in thrust rig per GE Drawing 246R702.

   b. Connect pressure transducer and valves per schematic GE Drawing 119C2652.

   c. Wire up valves, thermocouples, pressure transducer and power leads per GE Drawing 941D928.

      i. Thermocouple locations on thruster are shown on Drawing 47C141390.

3. Pressure check of gas system.

   a. With thruster solenoid valve de-energized, apply 100 psi of helium.

      i. Check systems for leaks.

      ii. Check output of pressure transducer.

      iii. Calibrate Sanborn recorder.

4. Check all electrical connections for continuity.

5. Calibrate thrust rig per Instruction I.

6. Close tank and evaluate to operating pressure of 1×10^{-5} mm Hg or better following procedure outlined in Section II-C.

B. Start-up and Operational Procedure for Thruster Operation and Monitoring Equipment

1. Veeco Vacuum Gauges Type RG-2A and RG-3A

   a. Type RG-3A

      i. Throw "Power On" switch to the up (on) position.

      ii. When "Microns of Mercury" meter reads less than "one" (full scale deflection) push "Filament on" button.

iii. With the "Pressure Multiplier" range set on "10^{-4}" throw "Read Current" switch to the up position. "Pressure mm Hg" meter should read approximately "1".

iv. In positions 10^{-5} and 10^{-6}, the meter should read full scale; if not, slowly rotate "Current Adjust" knob for full scale deflection.

v. As pressure decreases in the chamber it will be necessary to turn the "Pressure Multiplier" knob to the 10^{-5} range.

vi. Leave equipment on.

b. Type RG-2A

i. After reaching a vacuum of 10^{-4} mm Hg as indicated by the RG-3A Veeco gauge, push "Filament On" button.

ii. Complete steps 3 through 6 above.

iii. Leave equipment on.

2. Veeco TG-6 Thermocouple Gauge Control (2 Units)

a. Push "Power" button.

b. Pressure immediately registered on dial.

c. Leave equipment on.

3. RIDL Radiation Monitoring Equipment (2 Units)

a. Push "Power" buttons on left side of instrument rack. (Square above button will light).

b. Range knob will be set to values dictated by Health Physics personnel.

i. Set range knob to 10^5 disintegrations per minute full scale on log scale.

ii. 20 percent full scale background allowed.

iii. At 85 percent full scale, alarm triggers.

c. Leave equipment on.

4. Sanborn Console

a. Throw switch located on the top of the console to the on position.

b. Leave equipment on.

c. Equipment now ready for use.

d. Operating personnel have been "checked out" on the operations of this instrument.

5. Pulsing System

a. Throw "Power On" switch to the "On" position.

b. Set indicator dials to proper millisecond position.

c. Select single pulse or recurrent pulse as required.

d. Leave equipment on.

6. Daytronic Model DF 160 Displacement Transducer, Type 300C—Transducer Amplifier—Indicator

a. Throw power switch to "On".

b. Set range selector to "Zero".

c. Adjust amplifier zero to read exactly "Zero".

d. Set range selector switch to calibrate.

e. Adjust sensitivity to read full scale.

f. Turn range selector switch to "0.100" position.

g. Adjust transducer zero to read exactly "Zero".

h. Leave equipment on.

C. Normal Vacuum Start-up Procedure

1. All switches are in the down (off) position.

2. Push "Exhaust Fan" button.

3. Open manual hand valve for glove evacuation (Located on line to small Stokes pump).

4. Push "Small Stokes" button and "EP 1" toggle switch.

5. Push "Diffusion Pump" button, when pressure on TG-6 gauges is 100 microns or less.

6. Push "Stokes Mechanical Pumps" button.

7. Throw "Stokes Sol" to the up (on) position.

8. Turn "Stokes Blower" fully clockwise.

9. Wait one-half hour for diffusion pump to heat.

10. When chamber pressure is less than 50 microns, throw switch "EP 1" to the off position.

11. Throw switch "EP 2" to the on position.

12. At a pressure less than 5×10^{-4} mm Hg, turn on baffle cooler switch located at bottom center of panel.

13. With Isotope Start-Up Procedure

    With the promethium-147 capsule in the thruster chamber, these steps will be required after Step II. C. 2 in addition to those listed above

    2. (a) Switch "Scrubber" on

    2. (b) Switch on "Inert Gas" Ballast toggle switch

D. Power-Temperature Relationship Test

1. With heater de-energized and using helium propellant, adjust upstream pressure (P_o) so that chamber pressures over a range of 1 to 2

atmospheres will be obtained. A minimum of five points will be required to generate a suitable pressure ratio curve.

Read upstream pressure (P_o). Operate 3-way solenoid valve switch located under Daytronic Type 300 C Amplifier and read chamber pressure (P_c). Read thrust (F_o).

A pulse duration of 3 seconds should provide sufficient time to record both pressures. Points should be repeated twice to provide more data confidence.

Record all data on Sanborn. (Such data is recorded continuously.)

2. With heater de-energized and using ammonia propellant, adjust upstream pressure (P_o) so that chamber pressures over a range of 1 to 2 atmospheres will be obtained. A minimum of five points will be required to generate a suitable pressure ratio curve.

   Read upstream pressure (P_o). Operate 3-way solenoid valve switch and read chamber pressure (P_c). Read thrust (F_o).

   A pulse duration of 3 seconds should provide sufficient time to record both pressures. Points should be repeated twice to provide more data confidence.

   Record all data on Sanborn. (Such data is recorded continuously.)

3. Apply power in accordance with Instruction II to provide equilibrium temperature of 2000 °F*(T_1) and using helium propellant adjust upstream pressure (P_o) so that chamber pressures over a range of 1 to 2 atmospheres will be obtained. A minimum of five points will be required to generate a suitable pressure ratio curve.

   Read upstream pressure (P_o). Operated 3-way solenoid valve switch and read chamber pressure (P_c). Read thrust (F_o).

   A pulse duration of 3 seconds should provide sufficient time to record both pressures. Points should be repeated twice to provide more data confidence. Record all data on Sanborn. (Such data is recorded continuously.)

*2000 °F or maximum temperature achieved with same power output as fueled radioisojet thruster.

4. Apply power in accordance with Instruction II to provide equilibrium temperature of 2000 °F* (T_1) and using ammonia propellant adjust upstream pressure (P_o) so that chamber pressures over a range of 1 to 2 atmospheres will be obtained. A minimum of five points will be required to generate a suitable pressure ratio curve.

   a. Read upstream pressure (P_o). Operated 3-way solenoid valve switch and read chamber pressure (P_c). Read thrust (F_o).

      A pulse duration of 3 seconds should provide sufficient time to record both pressures. Points should be repeated twice to provide more data confidence.

      Record all data on Sanborn. (Such data is recorded continuously.)

5. Bleed nitrogen into the vacuum tank by throwing "Inert Gas Chamber" switch and adjusting fill valve on main pump port side of vacuum tank to provide pressure at 1×10^{-3} mm Hg. Allow temperatures to stabilize out and make steady state readings of all temperatures.

6. Bleed nitrogen into the vacuum tank to provide pressure of 1 atmosphere (or slightly less). Allow temperature to stabilize out and make steady state readings of all temperatures.

   NOTE: Care must be taken to zero the thrust rig output prior to each set of thrust readings in accordance with thrust rig calibration Instruction I.

E. Temperature Recovery Test

1. With the thruster stabilized out at 2000 °F* and supply pressure (P_o) adjusted to provide one and one half atmospheres chamber pressure (P_c) with ammonia, pulse the gas in accordance with the schedule shown below:

   a. 1.0 sec on, 1.0 sec off for five minutes,

   b. 1.0 sec on, 50.0 sec off until temperature T_1 comes to equilibrium,

   c. 1.0 sec on, 200.0 sec off until temperature T_1 comes to equilibrium,

*2000 °F or maximum temperature achieved with same power output as fueled radioisojet thruster.

d. 5.0 sec on, 200.0 sec off until temperature T_1 comes to equilibrium,

e. 30.0 sec on, 200.0 sec off until temperature T_1 comes to equilibrium,

f. 120.0 sec on, 200.0 sec off until temperature T_1 comes to equilibrium.

2. Data to be obtained.

   a. Measure thrust (F_o), heater element current (I_1), and voltage (V_1) continuously on Sanborn.

   b. All temperature to be continuously recorded.

F. Base Line Fuel Flow Test

1. Set power level to provide equilibrium temperature of 2000 °F and with ammonia supply pressure (P_o) set to provide one and one half atmosphere chamber pressure (P_c):

   a. Open the propellant valve and allow gas to flow until heater core temperature (T_1) reaches equilibrium (a change of 5 °F maximum over a period of 60 minutes).

   b. Record all temperatures (T) and thrust (F_o) as a function of time.

   c. Record heater element voltage (V_1) and current (I_1). Note that the power input should be maintained as constant as possible during this fuel flow test to simulate radioisotope heater.

G. Thruster Cool-Down Test

1. With thruster initially at 2000 °F*, shut off power and record cooling curve continuously until thruster temperature, T, equals less than 212 °F (100 °C). Measure time to cool down to 1400 °F, 1000 °F, 700 °F, 500 °F and 300 °F.

Measurement	Characteristic	Anticipated Range
P_o	Orifice pressure	0–10 atmospheres
P_c	Chamber pressure	0–2 atmospheres

*2000 °F or maximum temperature achieved with same power output as fueled radioisojet thruster.

Measurement	Characteristic	Anticipated Range
T_1	Core temperature	0-2000 °F
T_2 - T_{14}	Body temperature	0-700 °F
V_1	Heater voltage	0-30 volts
I_1	Heater current	0-4 amps
F_o	Thrust	0-100 mlb

Informative documents to be used with Test Procedure for Electrical Thruster:

GE Drawing 246R711 Assembly of Thruster
GE Drawing 246R702 Thruster and Thrust Rig
Drawing 119C2652 Gas Tube Connections, Thruster, Valves and Pressure Transducer
Drawing 941D928 Wiring Diagram (two sheets)
Drawing 47C141390 Thermocouple Locations on Thruster
Instruction I, Transducer and Thrust Rig Calibration
Instruction II, Electrical Thruster Heatup Procedure

H. General Vacuum System—Shut-Down Procedure

1. Throw "EP 2" switch to the off position.
2. Verify "EP 1" switch in the off position.
3. Close manual hand valve in glove port evacuation line.
4. Throw "INERT GAS CHAMBER" to the on position.
5. Wait for chamber to reach ambient pressure.
6. Open manual hand valve to let glove ports fill with air.
7. Glove ports may now be removed and work accomplished in thruster chamber.

Alternate Shut-Down Procedure (for extended shut-down periods)

1. Throw "EP 2" switch to the off position.
2. Throw baffle cooler switch to the off position.
3. Push "Diffusion Pump" button and allow to cool for one half hour.
4. Throw "EP 1" switch to the off position.
5. Close manual hand valve in glove port evacuation line.
6. Push "Small Stokes" button.
7. Turn "Stokes Blower" fully counter-clockwise.
8. Push "Stokes Mechanical Pump" button.
9. Throw "Inert Gas Chamber" to the on position.
10. Wait for chamber to reach ambient pressure.
11. Open manual hand valve to let glove ports fill with air.
12. Glove ports may be opened now. Also, if desired, the top half of the thruster test chamber may be lifted for access into the thruster area.

III. RADIOISOTOPE THRUSTER TESTS

A. Promethium-147 Fueled Radioisotope Capsule Handling

1. General Handling

   a. Shipping Cask: The shipping cask will be two concentric cylinders, both airtight. The outer cylinder will be a gamma shielding member plus a secondary physical barrier to possible escaped radioisotope. The inner cask (of stainless steel) will be the primary radioisotope capsule's physical barrier which will facilitate introduction of the promethium-147 capsule into the chamber in a safe manner.

   The shipping container must be equipped with two protected valves in order that a sample of the air inside the container can be

collected before the shipping container is opened. This safeguards against contaminating the test facility needlessly, should there have been a capsule failure.

The inner shipping cask will be filled with purified helium prior to shipping. This procedure will assure a clean ambient for the transfer of the capsule to Mound Laboratory.

b. The fueled capsule will be received and transferred to T-61 (at Mound) with Health Physics supervision. While the sub-assembly is still in the cask, an inert gas will be flushed through the cask through the needle valves and an air-borne beta analysis will be made. Also, "wipes" of the cask lid will be made per Health Physics practices.

c. If tests outlined above in III-A-1-b are negative, the inner cask will be lifted from the shipping container and placed on the shelf in the thruster chamber.

B. Installation of Radioisotope Capsule

1. Evacuate vacuum chamber to 1×10^{-5} mm Hg or best vacuum attainable in 2 hours following procedure outlined in Section II-C.

2. Backfill vacuum chamber with inert gas to one atmosphere (or slightly less) following vacuum system shut down procedure described in Section II-H.

3. Open up cask and, under the inert gas atmosphere, transfer the fueled capsule into the thruster sub-assembly using the special tools provided. Clamp capsule assembly in place and tighten wing nut.

4. Connect valve and orifice adapter to propellant connector on rear of thruster flange. Connect 1/16" diameter pressure sensor line. Connect thermocouple plugs to jacks mounted on base plate of thrust rig.

5. Calibrate thrust rig through gloves using procedure outlined in Instructions I and/or III.

6. Close glove port covers and evacuate vacuum chamber as per Section II-C.

7. Fueled tests now ready to begin.

C. Radioisotope Experimental Thruster Test Procedure

1. Constant Temperature Measurement Test

   a. Allow thruster temperature (T_1) to come to equilibrium (no more than 5°F change in 60 minutes).

   b. Adjust helium upstream pressure (P_o) with the regulator so that chamber pressures (P_c) of 5, 15, 20, 22 1/2, and 30 psi will be obtained. Use this data to generate a pressure ratio (P_o/P_c) curve. Pressure ratio data obtained during tests of electrically heated thruster can be used as a guide in setting supply pressure (P_o) for this test.

      i. A pulse duration of 3 seconds should be sufficiently long enough to enable both pressure readings to be made. Thrust readings (F_o) will also be made during these pulses. These points should be repeated twice to provide more data confidence.

      ii. Record core thermocouple reading (T_{33}) on Sanborn Model 350, Channel 3. Thrust (F_o) will be recorded on Channel 4. All other temperatures will be recorded on the two 0-1200°C L&N Speedomax Model W Strip Chart recorders.

   c. (Repeat—Substitute ammonia for helium)

2. Temperature Recovery Test

   a. Allow thruster temperature (T_1) to come to equilibrium (no more than 5 °F change in 60 minutes).

   b. Adjust ammonia propellant supply pressure (P_o) as measured at the orifice to provide 1.0 atmosphere thruster chamber pressure (P_c). Data obtained during electrical thruster tests will be useful in providing correlation between P_c and P_o.

   c. Pulse the gas in accordance with the schedule shown below. Pulse duration and pulse intervals are set on Tektronix-type 161 pulse generator and type 162 waveform generator. When pulses are longer than available automatically through this equipment, they will be operated manually by operating the appropriate selector switch.

i. 1.0 sec on, 1.0 sec off for 5 minutes.

ii. 1.0 sec on, 50.0 sec off until temperature comes to equilibrium.

iii. 1.0 sec on, 200.0 sec off until temperature comes to equilibrium.

iv. 5.0 sec on, 200.0 sec off until temperature comes to equilibrium.

v. 30.0 sec on, 200.0 sec off until temperature comes to equilibrium.

vi. 120.0 sec on, 200.0 sec off until temperature comes to equilibrium.

d. (Repeat b and c using 1.5 atmospheres instead of 1.0 atmospheres.)

e. (Repeat b and c using 2.0 atmospheres instead of 1.0 atmospheres.)

f. Measure thrust (F_o), core temperature (T_{33}), and orifice pressure (P_o) continuously on Sanborn strip chart recorder. All other thermocouples to be recorded continuously on the two 0-1200°C L&N Speedomax, type W strip chart recorders.

3. Base Line Full Flow Test

a. With thruster temperature (T_1) at equilibrium and with ammonia supply pressure (P_o) set to provide 1 atmosphere chamber pressure (P_c):

i. With propellant flow control selector switch in manual position, open the propellant valve and allow gas to flow until thruster core (T_1) reaches an equilibrium temperature (a change of 5 °F max over a period of 60 minutes).

ii. Record all temperatures (T_1-T_4) and thrust (F_o) as a function of time.

b. With thruster temperature (T_1) at equilibrium and with ammonia supply pressure (P_o) set to provide 1 1/2 atmospheres chamber pressure (P_c): Repeat steps i and ii.

c. With thruster temperature (T_1) at equilibrium and with ammonia supply pressure (P_o) set to provide 2 atmospheres chamber pressure (P_c): Repeat steps i and ii.

RANGES OF MEASUREMENTS

Measurement	Characteristic	Anticipated Range
P_o	Orifice pressure	0-10 atm.
P_c	Chamber pressure	0-2 atm.
T_1	Core temperature	0-2000°F
T_2-T_{14}	Body temperatures	0-700°F
F_o	Thrust	0-100 mlb.

Informative documents to be used with test procedure for radioisojet thruster:

Drawing 246R702 Thruster installed on thrust rig
Drawing 119C2652 Gas tube connections, thruster, valves, and pressure transducer
Drawing 47C141390 Thermocouple locations on thruster
Instruction I, Calibration of Thrust Rig

D. Re-casking of Fueled Radioisotope Capsule

1. Follow the applicable vacuum system shut-down procedure as outlined in Section II-H.

2. Remove the fueled capsule assembly from the heat shield. The following steps will be done:

   a. Lock the thrust rig by tightening the wingnut on locking screw.

   b. Disconnect the 1/8" fitting at the propellant and feed tube and the 1/16" "Swagelok" fitting at the pressure sensor. Also, unplug the mounting flange thermocouples from the jacks mounted on the thrust rig base plate.

   c. Loosen wing nut on the locking bar and slide out the fueled capsule assembly and insert into the inner cask.

d. Secure the shipping cask lid.

e. Health Physics personnel will make beta wipes of the inside of the chamber at this time. Their approval is required before opening the chamber to remove this inner cask and insert it into the outer shipping cask. This procedure is entirely up to the Health Physics Section.

f. If sealed assembly is deemed safe for removal, the shipping container is now ready to be transported to the transferral agency.

INSTRUCTION I

TRANSDUCER AND THRUST RIG CALIBRATION

With the table unlocked and the weight pan hanging in place, adjust the displacement transducer (DT) to mechanical and electrical zero according to Section C, parts 1, 2, and 3, "Initial Adjustment & Calibration, Using the Type 60 Plug-in Unit" of the Instruction Manual.

NOTE: Sec. C. part 2. Phase selector switch should be set at position "C" for the DF 160 model transducer.

Modified Calibration Procedure:

Turn the range selector to zero and readjust amplifier zero (if necessary) to achieve exactly zero reading. Set the Sanborn pen position to zero.

Turn the range selector to the 0.100" position. Adjust the transducer Zero to read exactly zero. Put 80-100 millipounds in the pan. Adjust the sensitivity so that the indicator reads exactly the weight used (100 full scale). Turn the range selector to CAL and (without disturbing any other controls) adjust the CAL set control (rear panel) to achieve exactly full scale reading. Adjust the Sanborn Gain control so that the pen reads full scale. The Sanborn will now read full scale with the amplifier indicator at all times, corresponding to the range selector switch position. Lock the CAL set control. Turn the range selector switch to 0.100" and remove the weight from the pan. The indicator should return to exactly zero. If it does not, check the amplifier zero and transducer zero and again apply the weight and recalibrate. The system is now calibrated and ready for use.

This calibration is valid as long as nothing is done to change the sensitivity of the entire rig such as the addition or removal of wires of any weight.

The thrust rig is now ready for use according to Section D. Thrust measurements can be read directly in millipounds on the amplifier indicator meter and on the external recording device (Sanborn).

NOTE: Linearity check should be made after installation of the thrust rig. Procedure: After calibration, set the range selector to - 0.100" and add weights in increments of about 10 millipounds up to 100. Plotting the indicator reading (displacement) vs. weight should yield a straight line if the system is linear.

INSTRUCTION II

ELECTRICAL THRUSTER HEATUP PROCEDURE

1. Increase heater voltage such that current level in heater is approximately 2.0 amps for three minutes.

2. Increase current to 4.0 amps. Readjust voltage in 10 minute increments to maintain 4.0 amps maximum.

3. When temperature (T_1) approaches 2000°F, or the temperature associated with power level of radioisotope heater, adjust voltage in proper direction to maintain that power level.

4. When core temperature (T_1) varies no more than 5°F in one hour, equilibrium conditions have been reached.

INSTRUCTION III

ADDITIONAL PROCEDURE FOR ACCESS INTO CHAMBER WHEN FUELED THRUSTERS IS IN PLACE

It will be necessary to pass items into the thruster test chamber on several occasions. These include hexagonal Allen wrenches, thermocouples, radiation detectors and screwdrivers. To avoid the wide-spread contamination which would result should a capsule rupture occur, access is made in the following manner.

(1) From left-hand glove on thruster test chamber, remove glove clamp and plastic tape. (Work is done with all personnel wearing respirators and under guidance of Health Physics.)

(2) Slowly pull glove such that right side of glove cuff is nearly off. Health Physics personnel will monitor continually by taking beta "wipes".

(3) Glove is pulled off enough to allow the item to be passed into the chamber. A second person in the right glove will receive the item and set it on the table.

(4) Replace glove, re-tape the cuff and tighten screw-down clamp.

(5) Chamber now prepared for normal access.

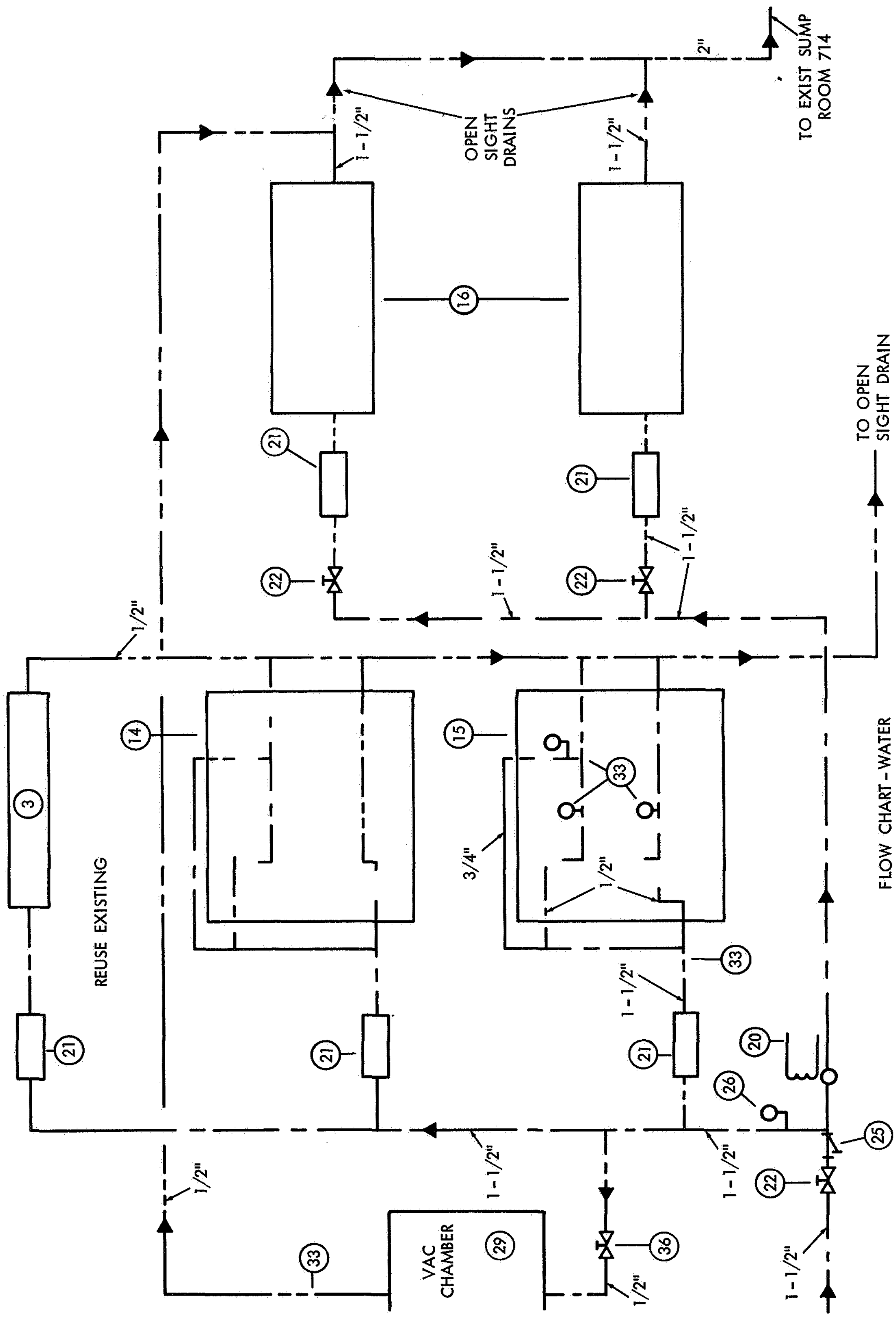
Mound Laboratory Flow Charts

LEGEND

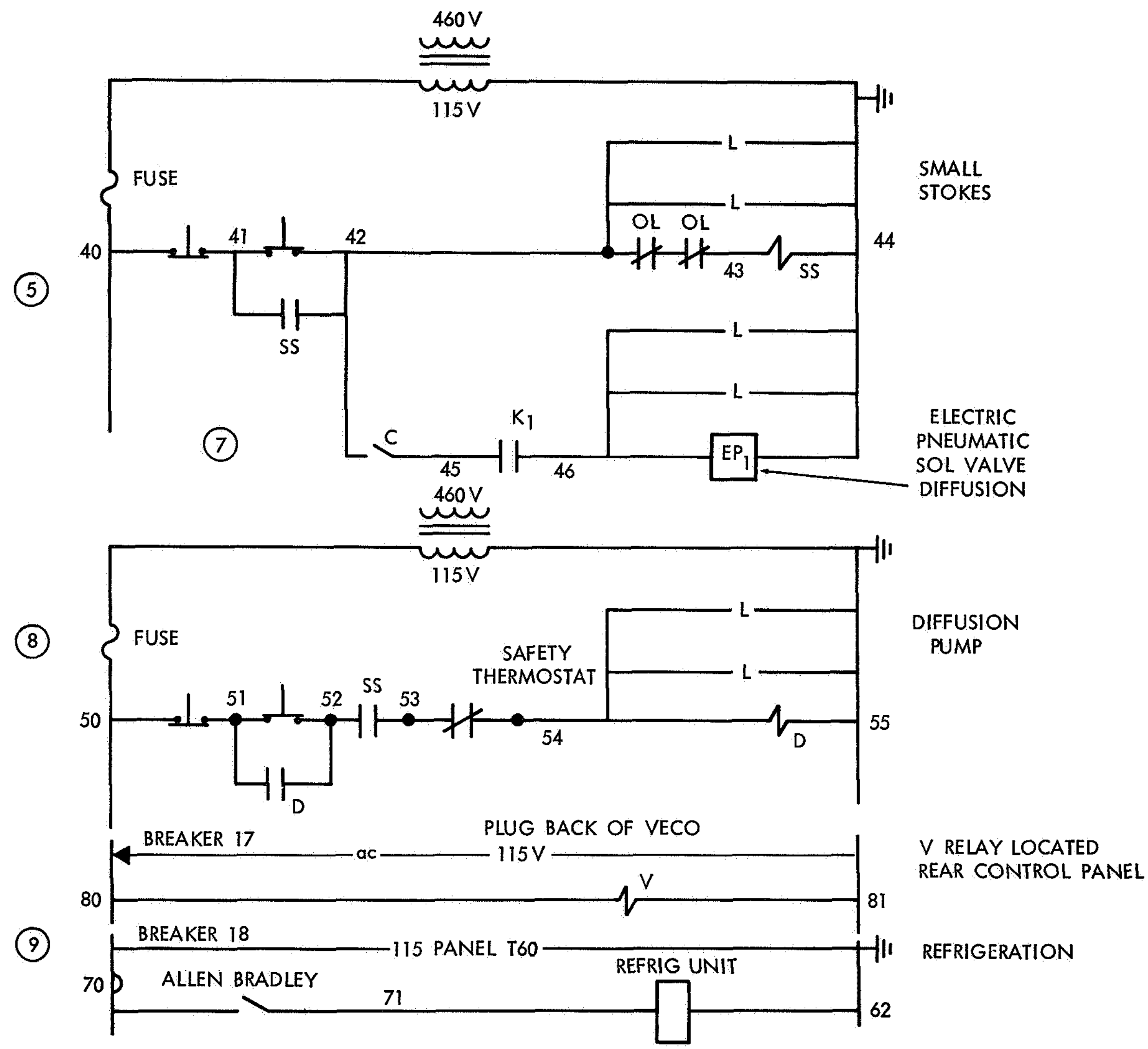
1 CYLINDERS - AMMONIA
2 MANIFOLD - 2 STATION
3 REGULATOR
4 REGULATOR - ANNIN
5 SOLENOID
6 DRIER
7 FILTER
8 S.O. VALVE - THROTTLE
9 THRUSTER
10 FILTER
11 CHECK VALVE
12 FLEXIBLE CONNECTION
13 DIFFUSION PUMP
14 STOKES - EXISTING
15 STOKES - NEW
16 SCRUBBERS
17 DEMISTER
18 FILTER
19 BLOWER - EXHAUST
20 SOLENOID - WATER
21 INDICATOR - FLOW
22 S.O. VALVE
23 S.O. VALVE
24 SOLENOID VALVE
25 CHECK VALVE
26 GAUGE
27 FITTING - SWEDGELOCK
28 VALVE - RELIEF
29 VACUUM CHAMBER - EXISTING
30 VALVE - VACUUM GATE
31 HOUSING
32 SAFETY MASK
33 THERMOMETER
34 PIPING
35 VALVE S.O.
36 REFRIGERATION UNIT
37 VALVE - STOCK NO. 7 - 6763
38 CYLINDER - HELIUM
39 REGULATOR
40 GAUGE - HEISE
41 SPORLAN CATCHALL
42 REFRIG SIGHT GLASS
43 A.P. REFRIG CONTROL
44 GASKETS
45 FILTER
46 METER
47 CHAMBER - BY GE
48 FILTER BY GE
49 SOLENOID - BLEED

CHANGES

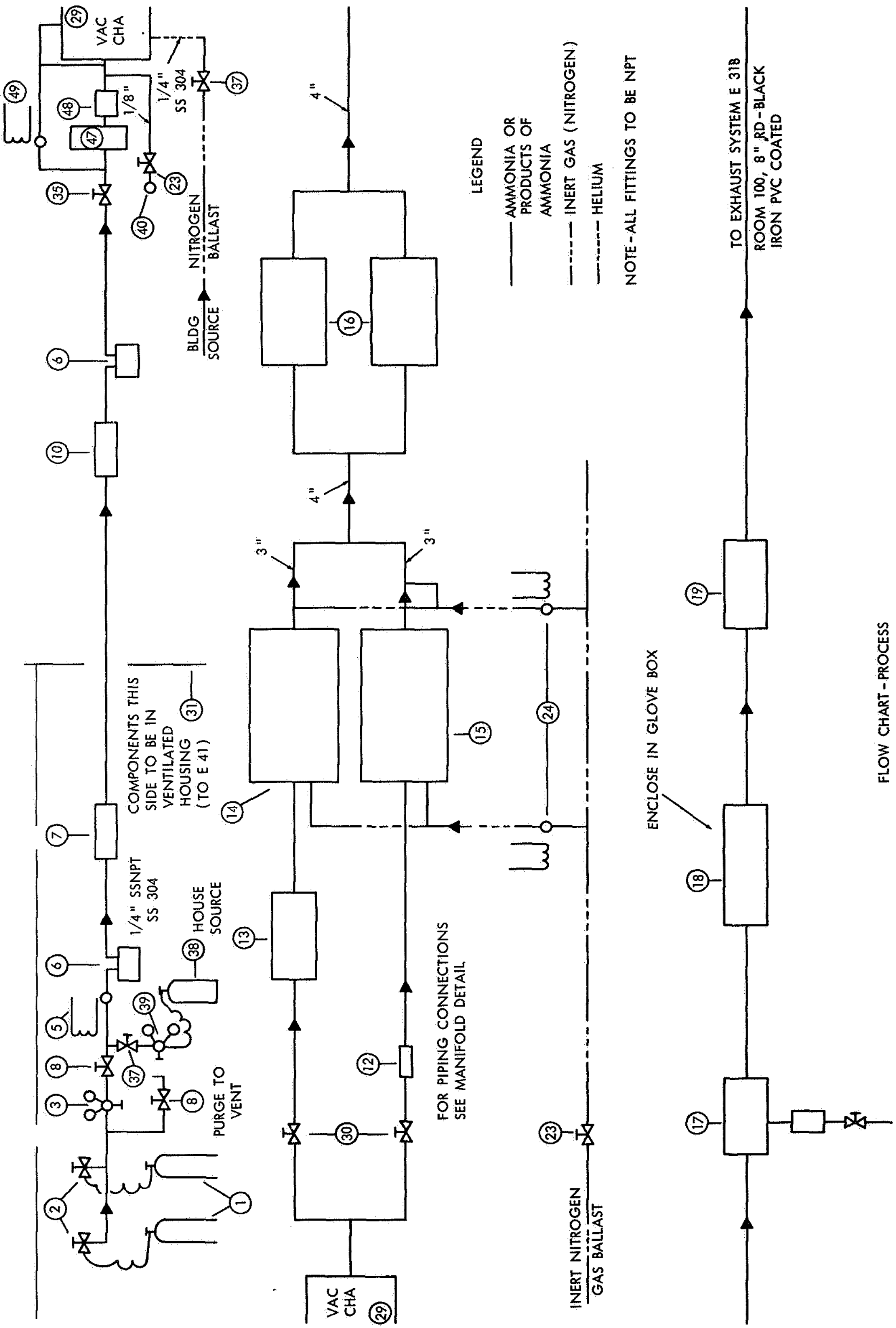
A DELETION THRUSTER
B EXHAUST SYSTEM E22 REDESIGNATED AS E 31B
C ADDITION OF HELIUM CIRCUIT, TO POODLE
D ADDITION OF WATER CIRCUIT TO POODLE
E ADDITION OF HELIUM CIRCUIT TO TEST
F ADDITION OF HEISE GAUGE

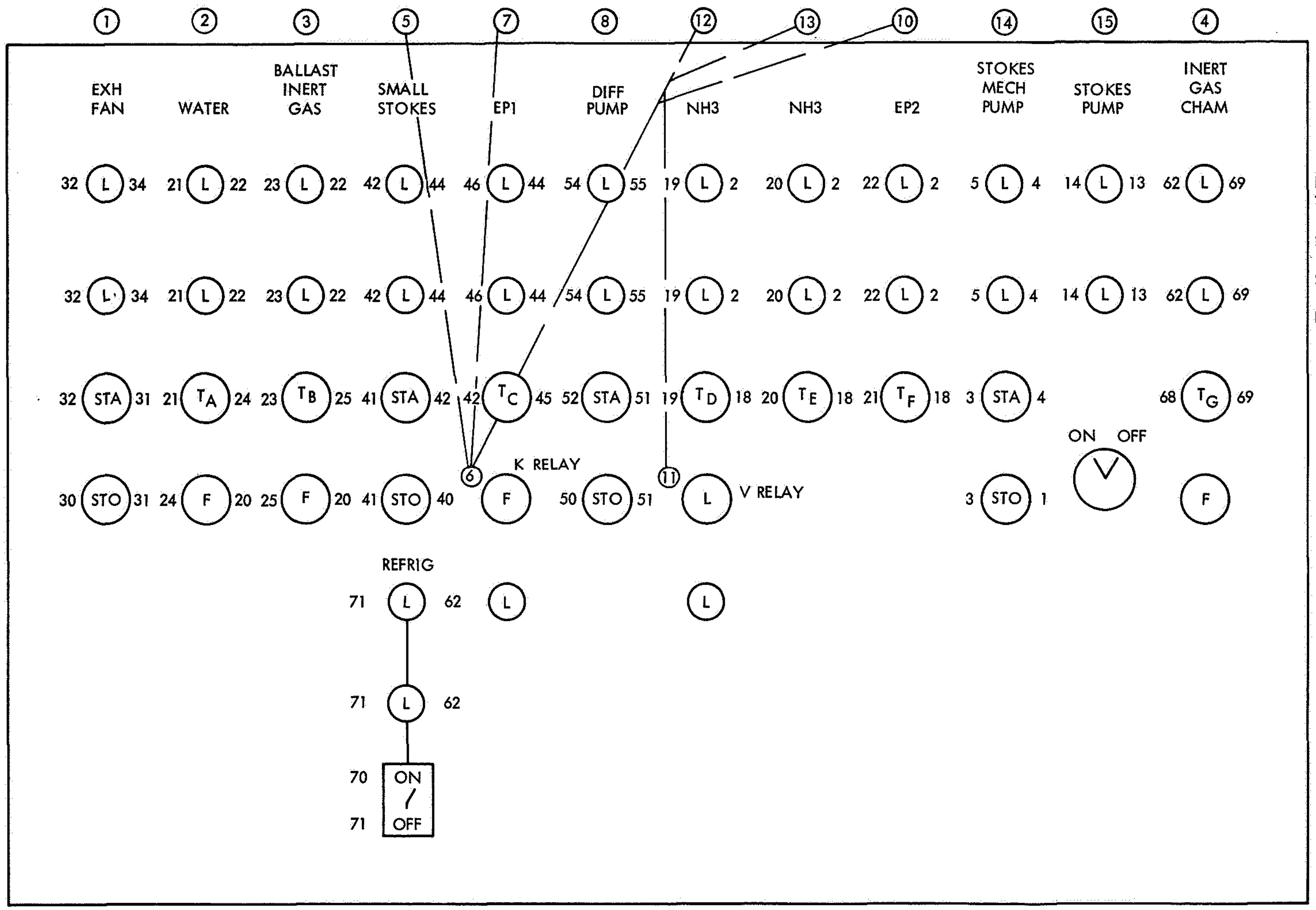


FLOW CHART-WATER



FLOW CHART
ELECTRICAL


VAC CHA
29
1/4" SS 304
1/8"
NITROGEN BALLAST
BLDG SOURCE
4"
3"
LEGEND
AMMONIA OR PRODUCTS OF AMMONIA
INERT GAS (NITROGEN)
HELIUM
NOTE - ALL FITTINGS TO BE NPT
COMPONENTS THIS SIDE TO BE IN VENTILATED HOUSING (TO E 41)
1/4" SSNPT SS 304
HOUSE SOURCE
PURGE TO VENT
FOR PIPING CONNECTIONS SEE MANIFOLD DETAIL
ENCLOSE IN GLOVE BOX
INERT NITROGEN GAS BALLAST
TO EXHAUST SYSTEM E 31B ROOM 100, 8" RD-BLACK IRON PVC COATED
FLOW CHART - PROCESS



L - LIGHT
T_A - TOGGLE SW1
STA - START MOM CON
STO - STOP MOM CON
F - FUSE 5 AMPS

NOTES - MOUNT 40 TERMINAL STRIP REAR OF CONTROL PANEL
MOUNT K RELAY BACK OF CONTROL PANEL. K RELAY INTERLOCKED WITH SPEED - MAX
MOUNT V RELAY BACK OF CONTROL PANEL. V RELAY ENERGIZED BY VECO INST

FLOW CHART – ELECTRICAL

APPENDIX C

TEST RESULTS

CONTENTS

APPENDIX C

TEST RESULTS

1. General Test Conditions:

Three gases were used in the Radioisojet demonstration test:

(a) Ammonia as a propellant gas,

(b) Helium as a propellant standard and also as a back-fill gas, and

(c) Nitrogen as a back-fill gas (which simulated the thermal conductivity of air).

The ammonia was Baker 99.99% minimum NH_3. Gas analysis showed the water concentration was less than 50 ppm (weight) and the oil was given as less than 3 ppm by weight. The helium and nitrogen were both from Matheson Scientific Company. Analyses given were 99.995% He and 99.997% N_2, respectively.

Propellant flow rate was determined by reading the differential pressure on a previously calibrated orifice. The calibration curve, "Pressure vs Mass Flow Rate," for helium and ammonia is shown in Figure C-1. This curve is based on an orifice temperature of 70°F. The orifice calibration was performed using a Precision Wet Test Meter, a water displacement, rotary type meter having a volume accuracy to within ± 1/2%. Flow rate connections for orifice temperature increase were made using the curve of Figure C-2.

The vacuum levels achieved at Mound under the flow-no flow conditions are as follows:

(a) No Flow Conditions

Under no flow conditions, the best vacuum observed was 3×10^{-6} torr as measured by the Veeco Ion Gauge. The filaments on these gauges proved unacceptable for this use as all four were burned out before the end of the test. After they were burned out, the thermocouple readings were relied upon for the pressure measurement.

(b) Flow Conditions

During ammonia flow with a thruster chamber pressure of approximately 100 psi (maximum flow rate), the Stokes 1719 pump system was able to maintain a pressure of less than 3×10^{-4} torr.

2. Electrical Simulator

In order to generate a Power vs Temperature curve, the heater power was lowered to 56.2 watts, allowed to stabilize overnight, then raised to 76.6 watts and again allowed to stabilize overnight. The power supply was found to be fluctuating at the 56.2 watt level so this point was not used. A later point taken immediately prior to the power off cool-down in vacuum was used for the lower temperature point. The Power vs Temperature curve is shown in Figure V-1 along with the initial curve generated at GE before delivery of the thruster. The post-fueled (electrical) thruster test data at Mound and GE is also included for convenience.

In conformance with the Standard Operating Procedure (Appendix B), the thruster was then pulsed, one second on, one second off for five minutes. Since one second was not enough time for the thrust rig to stabilize, no thrust measurements could be taken. Temperatures were recorded by reading the millivolt potentiometer since the Sanborn recorder trace was not reliable enough due to electronic drift and chart paper shifting.

In order to speed up the testing, it was decided to change the sequence of the cycle tests and perform them in the order of increasing duty cycle. By starting each cycle test immediately after the previous one, the base temperatures could be obtained without waiting for the temperature to recover.

Following the 1 second on, 1 second off cycle, the thruster was allowed to warm up to equilibrium. The next pulse mode was 1 second on, 208 seconds off. (The pulse generator output cycle was 208 seconds at the nominal 200 seconds setting.) The temperature drop after 42 minutes was only 9°F and since the power was fluctuating, the next pulse mode(5 seconds on, 208 seconds off) was begun. A line voltage regulator was installed in the power supply input during this test, but a steady drop in power was still observed. The core temperature dropped 16 degrees during this period of 33 minutes and the test was terminated to observe the power and warm-up. In 42 minutes the temperature rose 1°F while the power remained constant.

The remainder of the cycling tests were conducted somewhat hurriedly under the same conditions with the reservation that more electrical testing could be done after the fueled test.

The most consistent and significant data was obtained during the 120 seconds on, 200 seconds off pulse mode (Figure C-3) where readings were taken approximately every 30 seconds on the millivolt potentiometer. Once again the power was dropping so that the overall temperature decrease, due to the pulsing, could

not be determined. The temperature fluctuations, however, are quite evident. After 42 minutes, the thruster valve was left open for the continuous flow mode. Again, temperatures were being read on the millivolt potentiometer every 30 - 60 seconds (Figure C-3). After 34 minutes of this, it was decided to cut the thermocouple lead and connect it to the 12 point Speed-O-Max recorder in place of one of the unnecessary thermocouples. (TC #3, Figure IV-4, was dropped as its maximum temperature was 65°F.) The propellant flow was cut off and the thruster allowed to warm up overnight. With the electronics again in order and the temperature stable, the heater power was turned off and the cool-down curve was obtained (Figure C-4).

Figure C-5 shows the Pressure Ratio (Po/Pc) versus Core Temperature, using data obtained during the pulse modes, with respect to the theoretical ratio expected.

3. Promethium Fueled Thruster Testing

The temperature drop versus time for each of the pulse modes recorded during the cycle tests is illustrated in the series of figures at the end of this Appendix in the order outlined. Temperature variation due to the individual pulses is also included where significant. The band width covers the maximum and minimum temperature measured during the individual pulses. The insert on the respective graphs is an actual copy of the strip chart used to record the temperature. The section used is indicated by a dotted rectangle on the complete curve. Included in the series, after the pulse data curves, are two curves illustrating the effect of backfilling the vacuum chamber to one atmosphere of nitrogen and helium respectively. With the vacuum system backfilled with helium, a steady state temperature of 560°F was measured. The nitrogen atmosphere, which was chosen to simulate conditions a thruster might encounter on a launch pad, allowed the temperature to drop to only 965°F as measured by the heater core thermocouple.

4. Incidents

a. Thermocouple Replacements. The radioisotope thermocouple could not be installed properly at Mound Laboratory. It was impossible to insert the cladded thermocouple into the well entirely without partial disassembly of the thruster. The dismantle was accomplished and a new thermocouple was installed. However, pyrometer readings were taken just prior to removal of the capsule from the heat shield at the conclusion of the fueled test sequence (at a vacuum of approximately 5 x 10^{-5} torr). The results, listed in Table C-1, would tend to substantiate the suspicion that the core thermocouple was not completely seated against the end of the capsule, or the results could indicate a thermal gradient.

Table C-1
Optical Pyrometer—Thermocouple Temperature Measurement Comparison

Optical Pyrometer (corrected)*	L&N Speed-O-Max Recorder**	L&N Millivolt Potentiometer**
865°C (1590°F)	795°C (1465°F)	790°C (1454°F)

*Pyro Micro-Optical Pyrometer was calibrated using an NBS tungsten strip lamp immediately after the measurement. The temperature was corrected for the absorptivity of the one-inch plexiglass window. The emissivity of the nozzle orifice was assumed to be unity. Vacuum was approximately 5×10^{-5} torr.

**Thermocouples (chromel-alumel) were the "core" thermocouples.

An additional thermocouple was affixed to the metering orifice. The measurement was taken for the purpose of correcting flow rate data. Flow rates were computed based on a propellant temperature measurement which was upstream of the orifice. During the test it was noted that the propellant temperature in the area of the orifice was different than that at the pick-up point. Therefore it was necessary to correct for the propellant density change.

b. EP-2 Blow-By Incident. On one occasion during thruster tests, valve EP-2 was opened before the Stokes 1719 Blower was in operation. The thruster test chamber was under high vacuum of less than one millitorr. The in-rushing gust of air destroyed the thin window Geiger-Mueller tubes, jarred the test stand hard enough to break a wire in the displacement transducer, dislodge the back-plate of the thruster and disperse throughout the chamber the 4 cm filter paper wipes.

The repairs were made as follows:

(1) A second diffusion-junction detector was installed as a Health Physics precaution.

(2) A new transducer pick-up was obtained and installed.

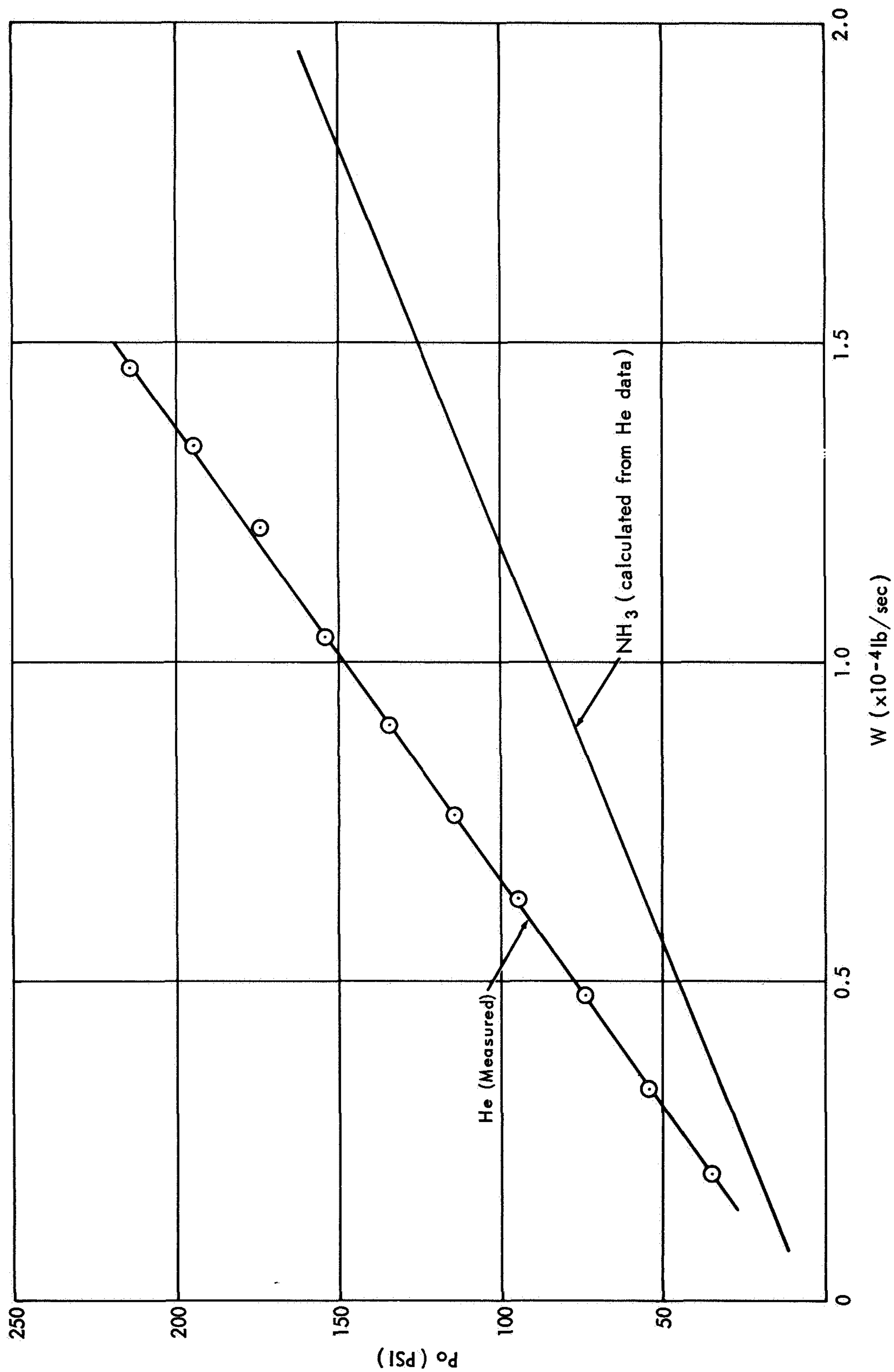
(3) The back-plate was taped back into its original position.

After this event, the no-flow equilibrium temperature was 1508°F, or essentially unchanged. Apparently little damage had been done to the radiation shield system.

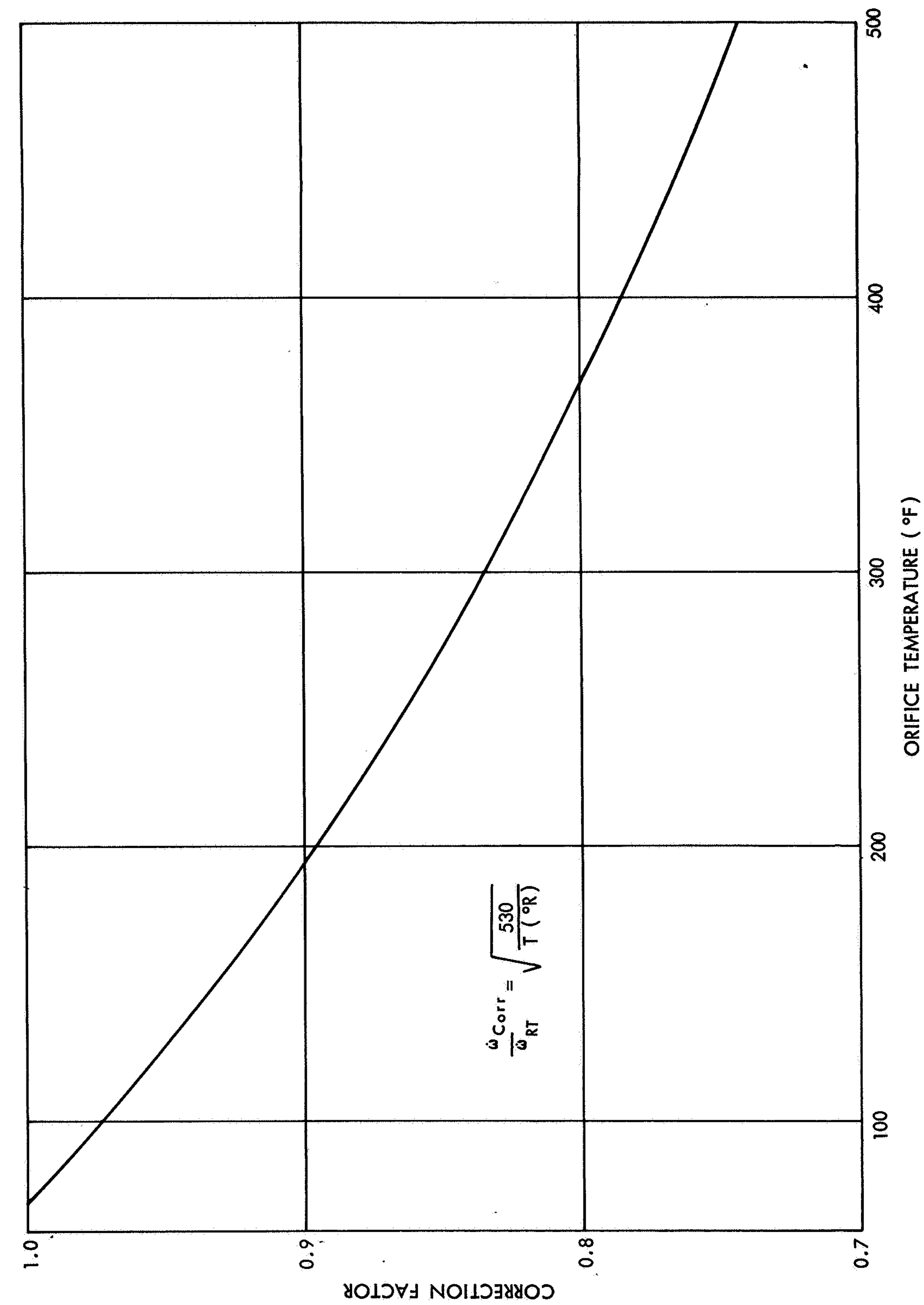
An interlock system was added which prevented the opening of EP-2 unless the Stokes 1719 was operating on the blower mode. This ensured that the incident could not be repeated.

5. Promethium Fueled Thruster Cycle Test Temperature Data

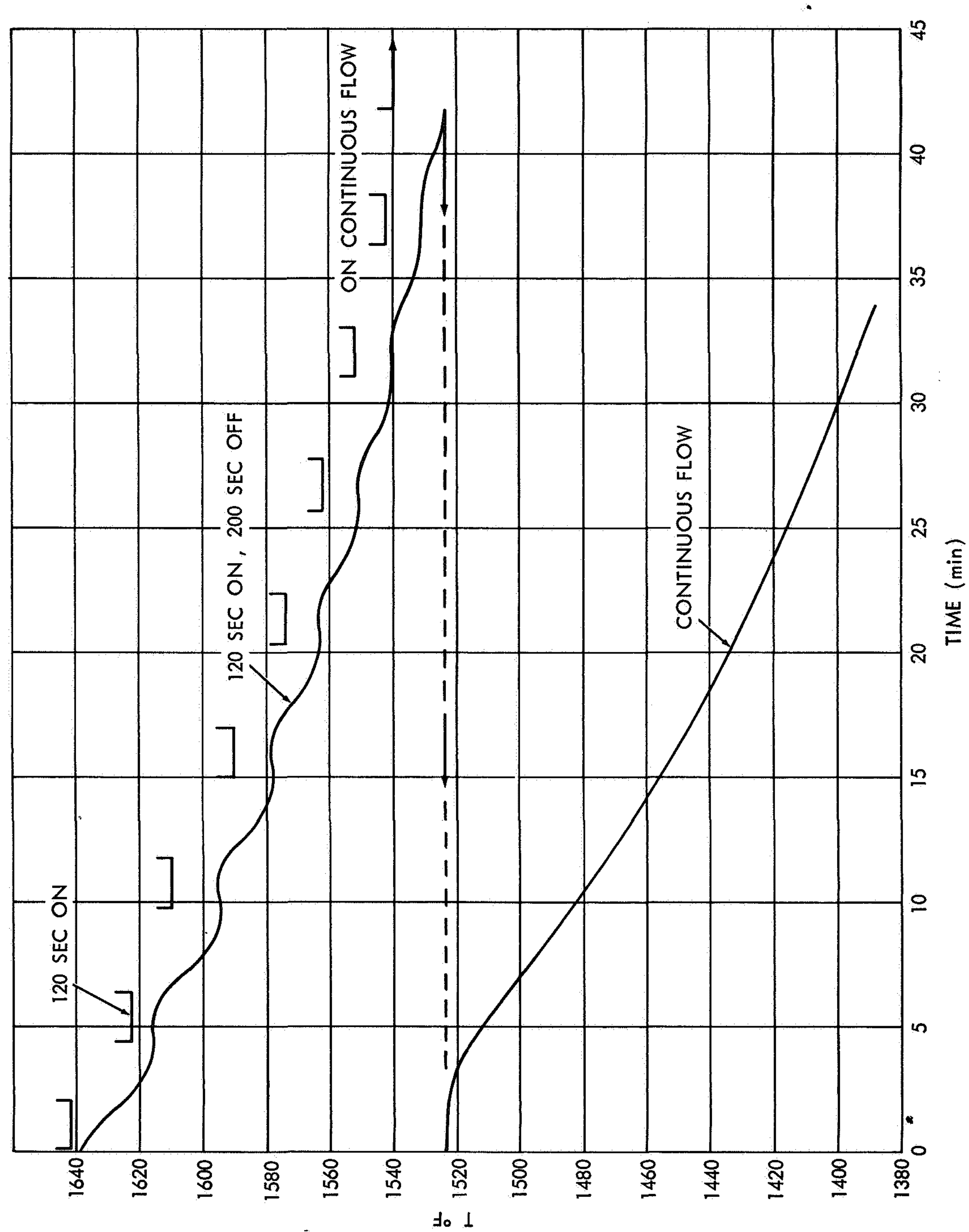
The following series of figures represent, in order, the fueled thruster cycle tests. Each curve consists of a plot of core temperature versus time.



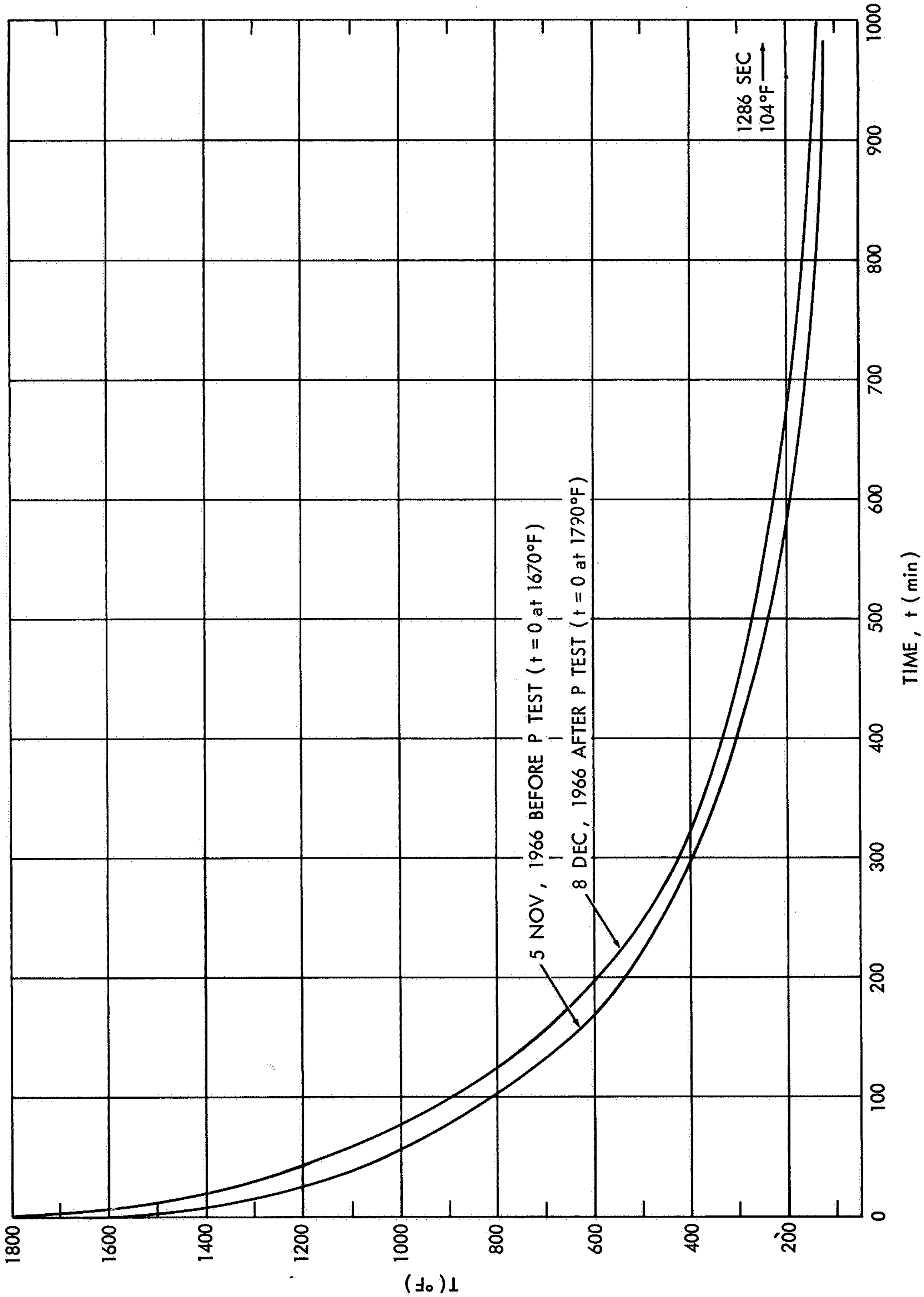
Po (PSI)
250
200
150
100
50
0
0.5
1.0
1.5
2.0
W (x10^-4 lb/sec)
He (Measured)
NH3 (calculated from He data)

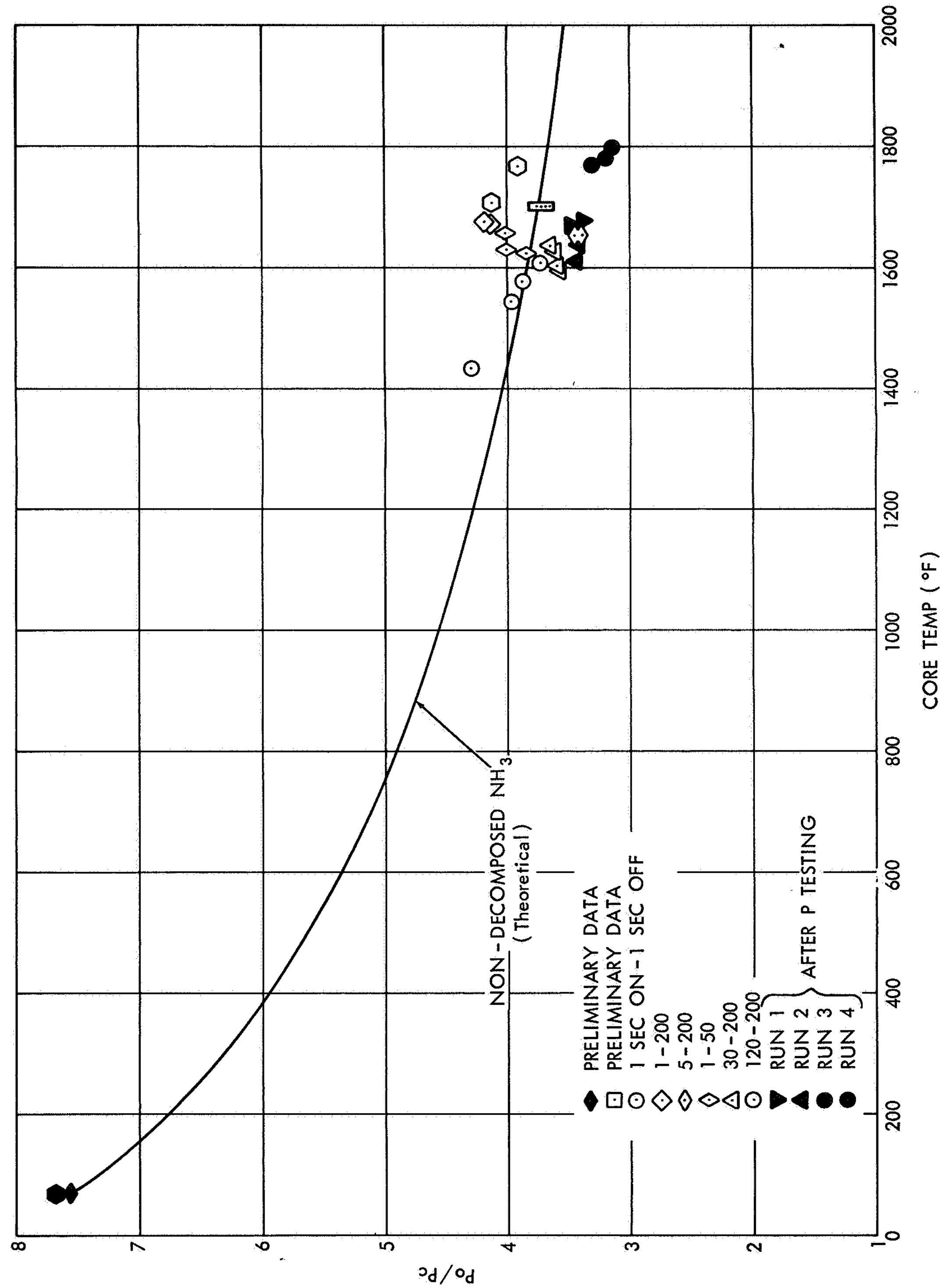


$\frac{\dot{\omega}_{Corr}}{\dot{\omega}_{RT}} = \sqrt{\frac{530}{T\ (°R)}}$
CORRECTION FACTOR
1.0
0.9
0.8
0.7
ORIFICE TEMPERATURE (°F)
100
200
300
400
500

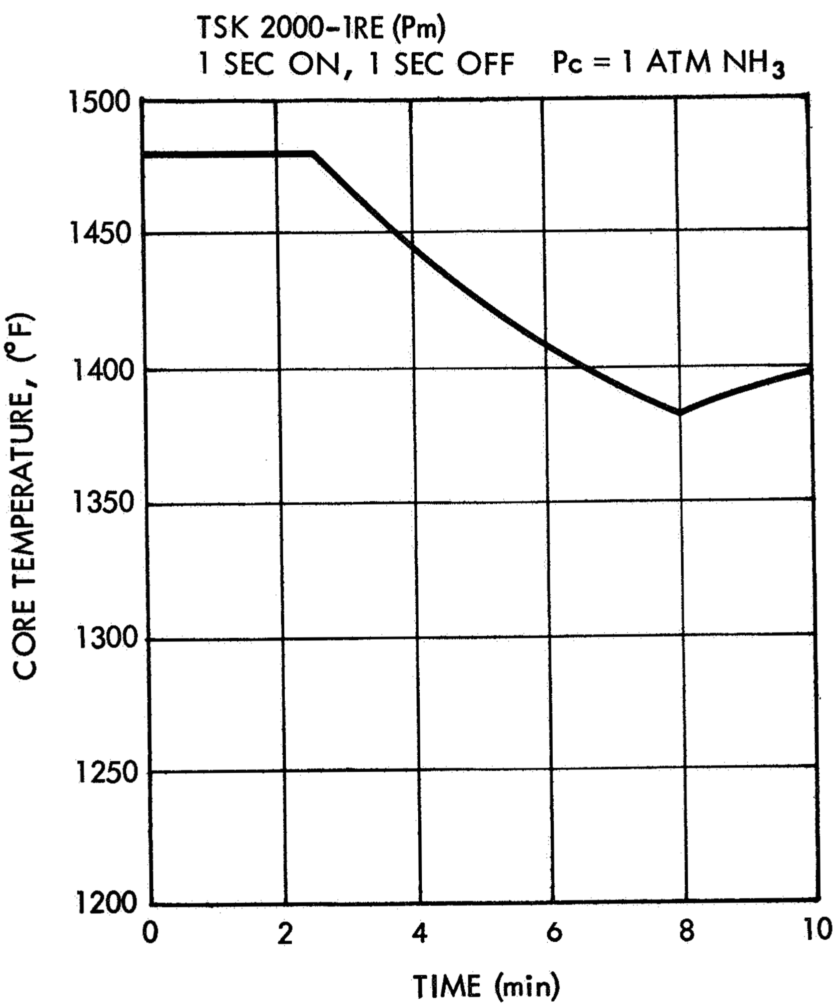


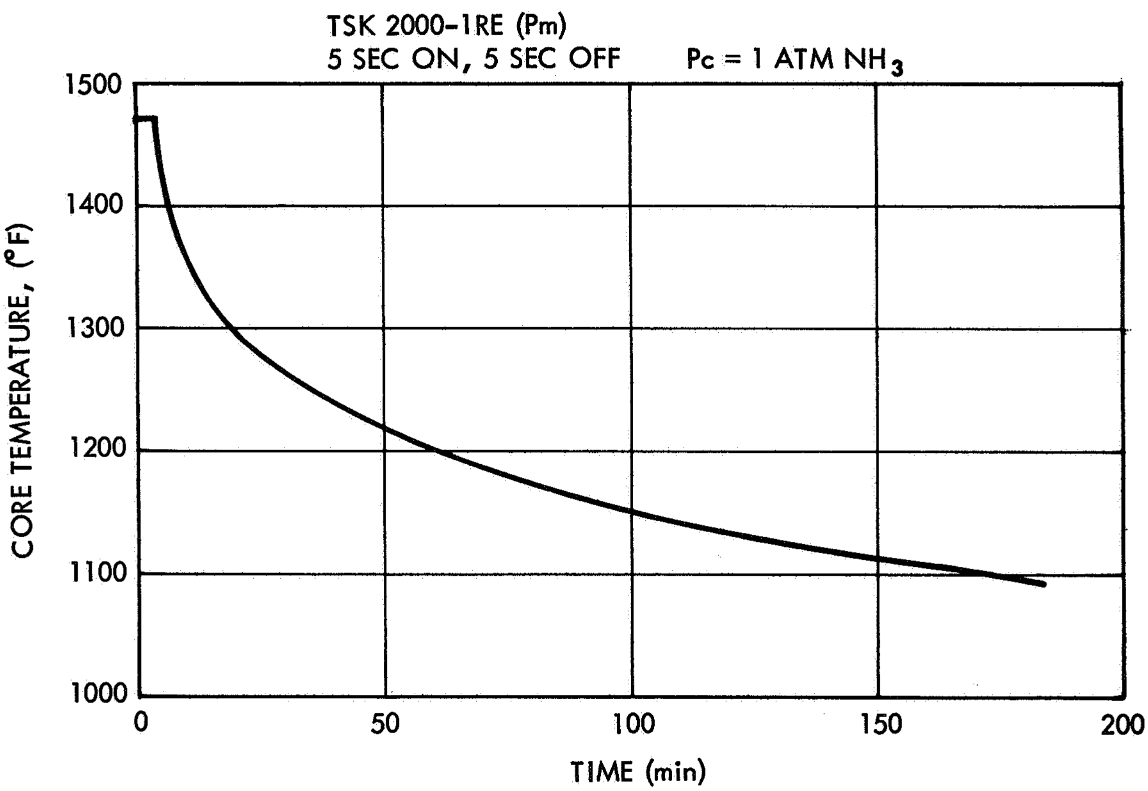
120 SEC ON
120 SEC ON , 200 SEC OFF
ON CONTINUOUS FLOW
CONTINUOUS FLOW
T °F
TIME (min)
1640
1620
1600
1580
1560
1540
1520
1500
1480
1460
1440
1420
1400
1380
0
5
10
15
20
25
30
35
40
45

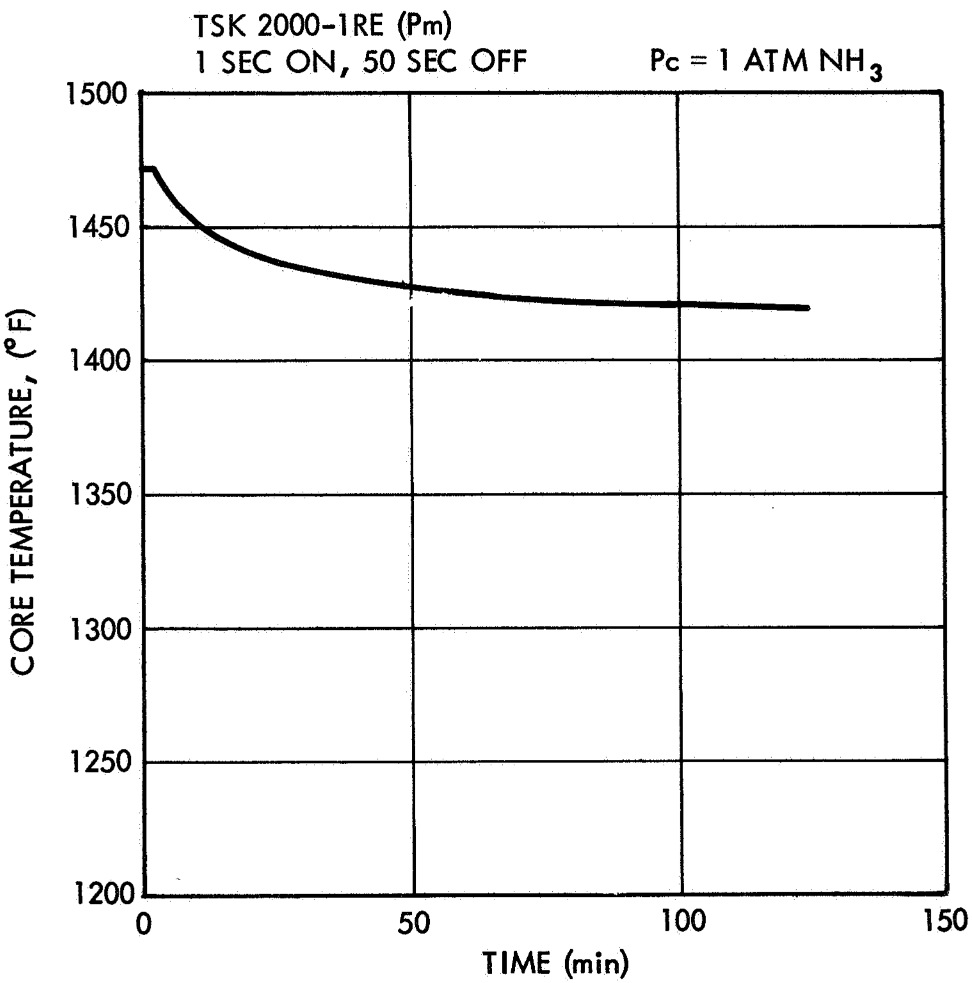

5 NOV, 1966 BEFORE P TEST (t = 0 at 1670°F)
8 DEC, 1966 AFTER P TEST (t = 0 at 1790°F)
1286 SEC
104°F
TIME, t (min)
T (°F)
0 100 200 300 400 500 600 700 800 900 1000
1800 1600 1400 1200 1000 800 600 400 200

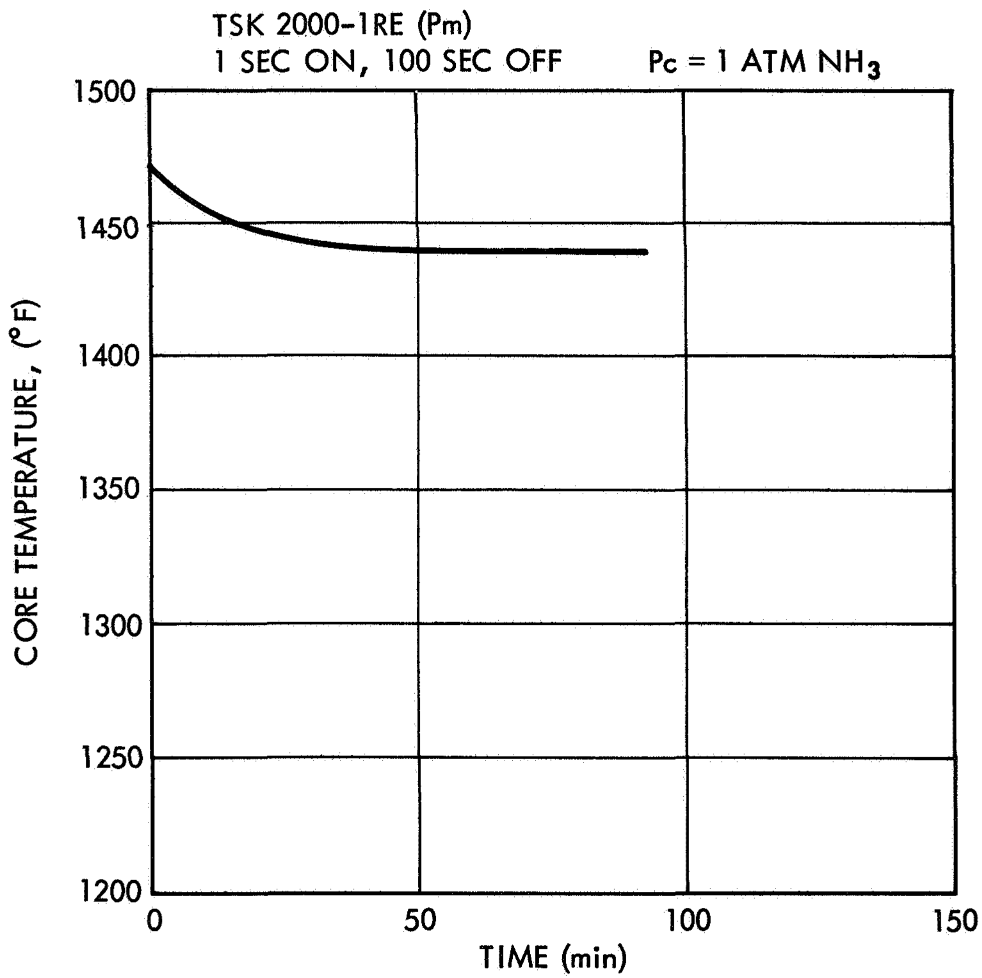


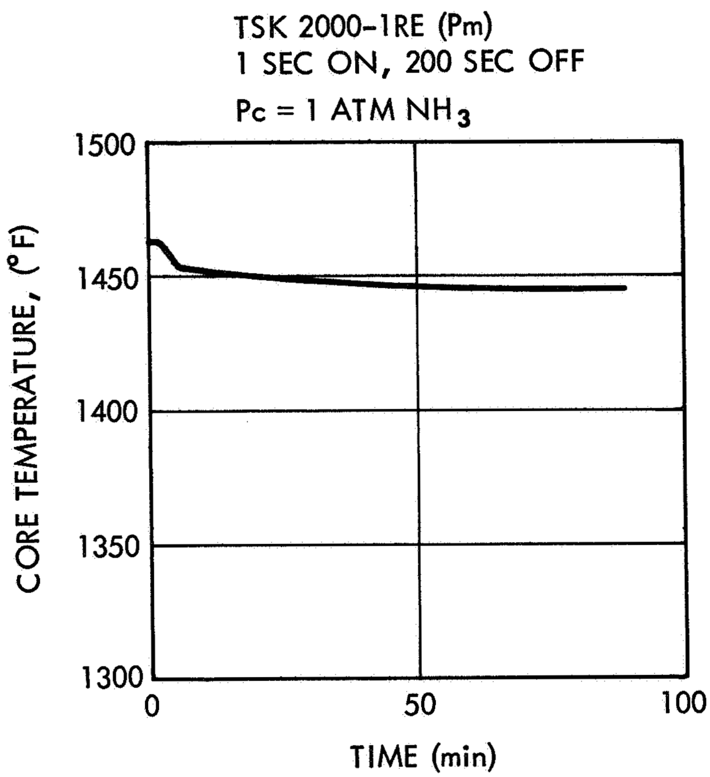
Po/Pc
8
7
6
5
4
3
2
1
0
200
400
600
800
1000
1200
1400
1600
1800
2000
CORE TEMP (°F)
NON-DECOMPOSED NH_3
(Theoretical)
PRELIMINARY DATA
PRELIMINARY DATA
1 SEC ON-1 SEC OFF
1-200
5-200
1-50
30-200
120-200
RUN 1
RUN 2
RUN 3
RUN 4
AFTER P TESTING

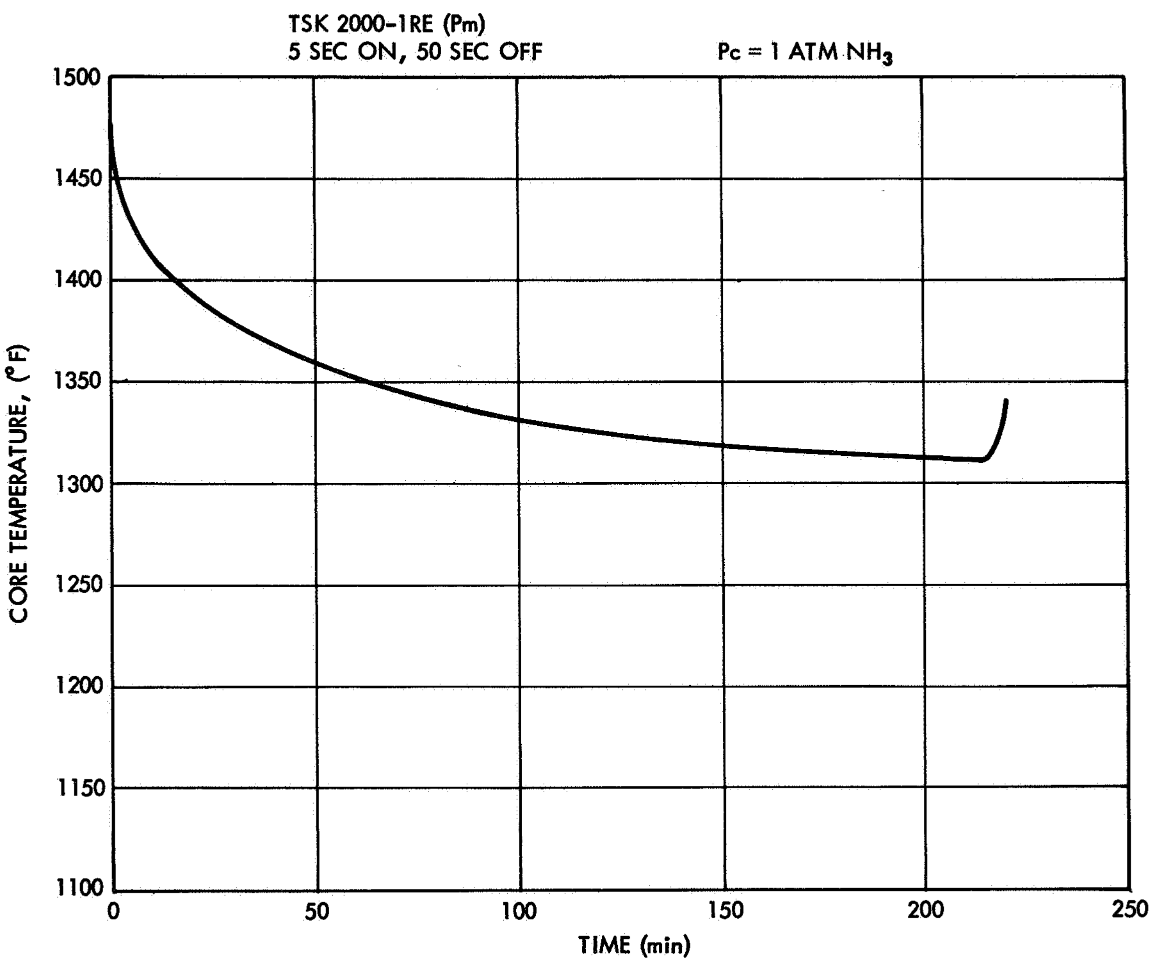

TSK 2000-1RE (Pm)
1 SEC ON, 1 SEC OFF Pc = 1 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
0
2
4
6
8
10
TIME (min)

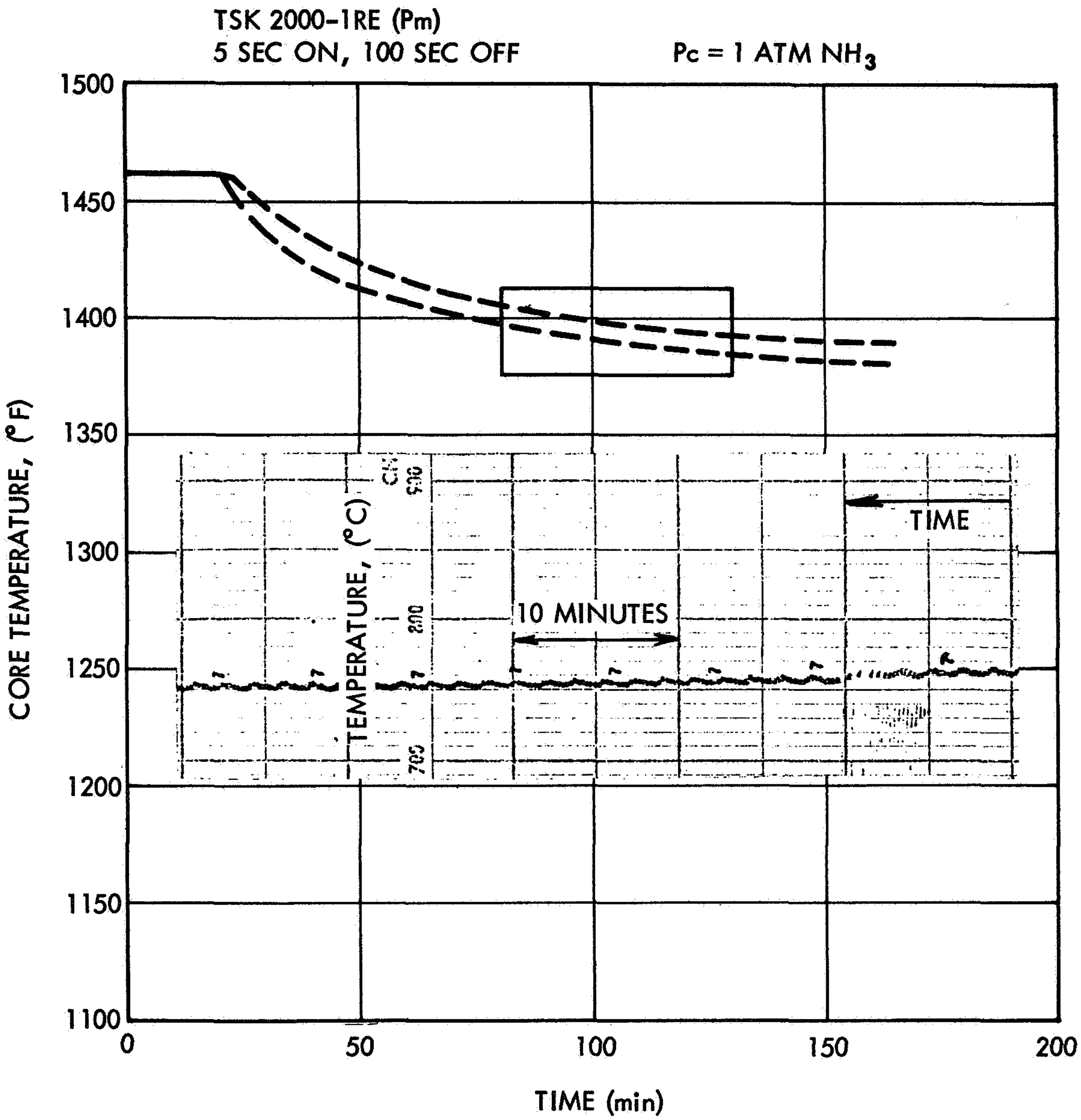

TSK 2000-1RE (Pm)
5 SEC ON, 5 SEC OFF
Pc = 1 ATM NH_3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
0
50
100
150
200
TIME (min)


TSK 2000-1RE (Pm)
1 SEC ON, 50 SEC OFF
Pc = 1 ATM NH_3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
0
50
100
150
TIME (min)

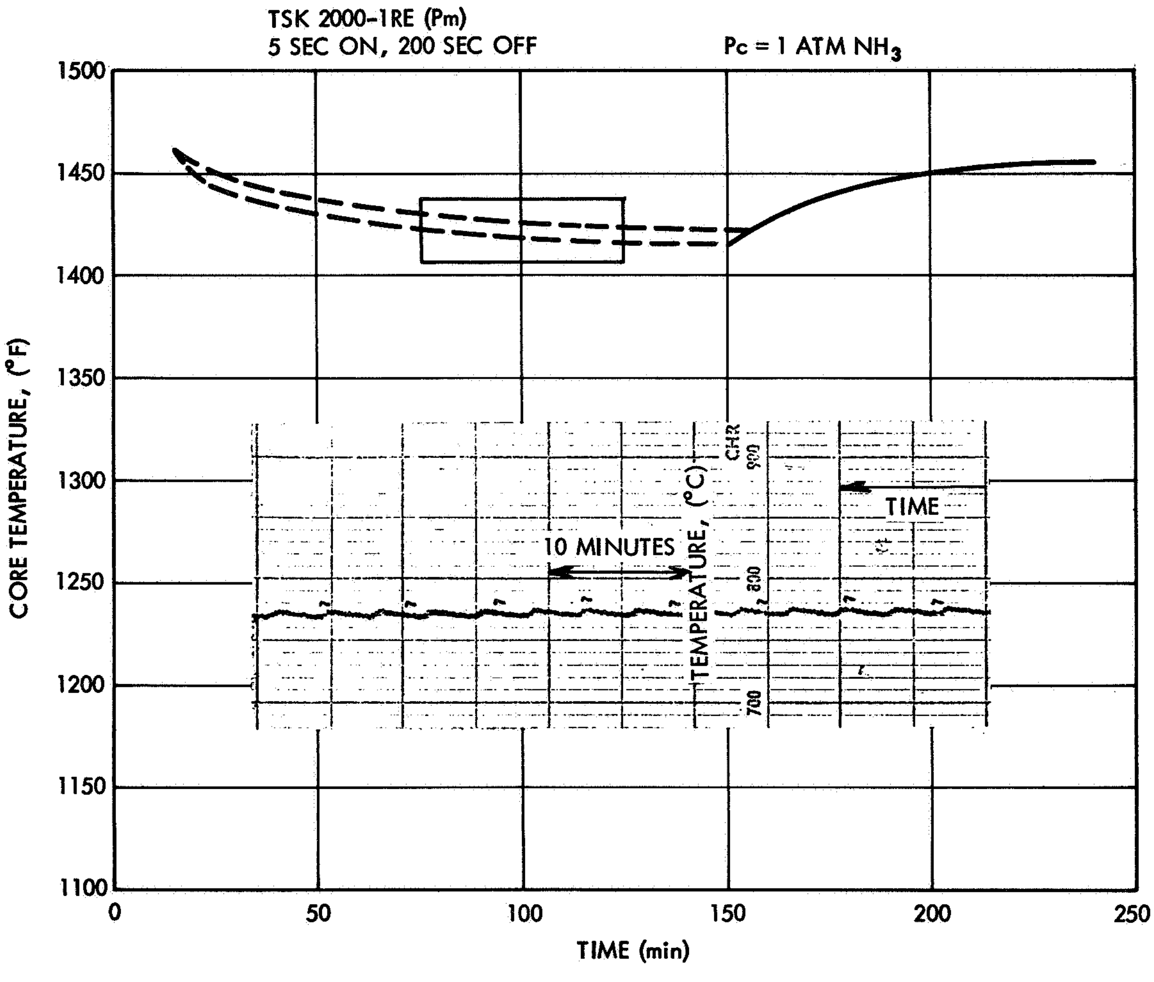

TSK 2000-1RE (Pm)
1 SEC ON, 100 SEC OFF
Pc = 1 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
0
50
100
150
TIME (min)

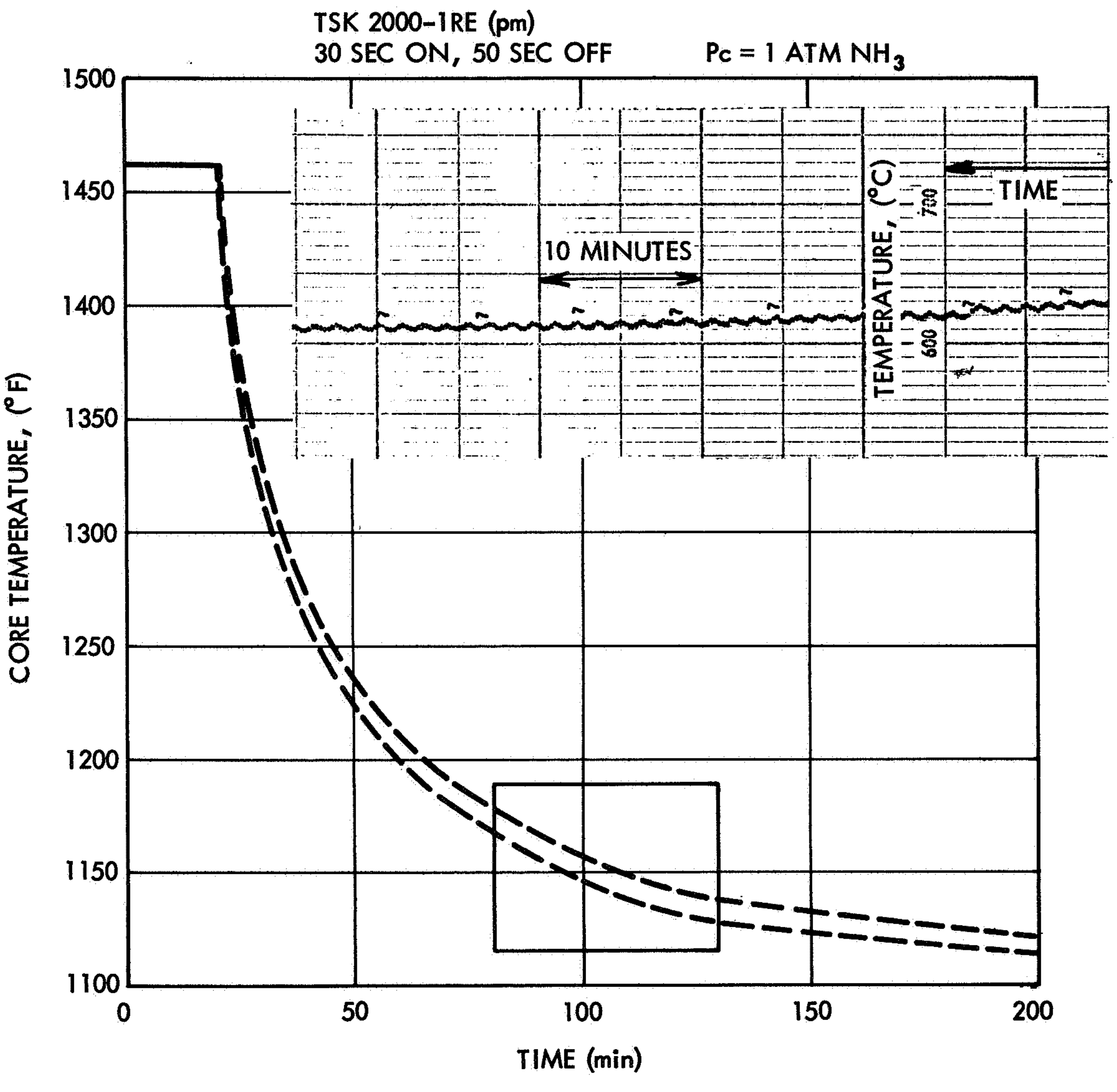

TSK 2000-1RE (Pm)
1 SEC ON, 200 SEC OFF
Pc = 1 ATM NH3
1500
1450
1400
1350
1300
CORE TEMPERATURE, (°F)
0
50
100
TIME (min)

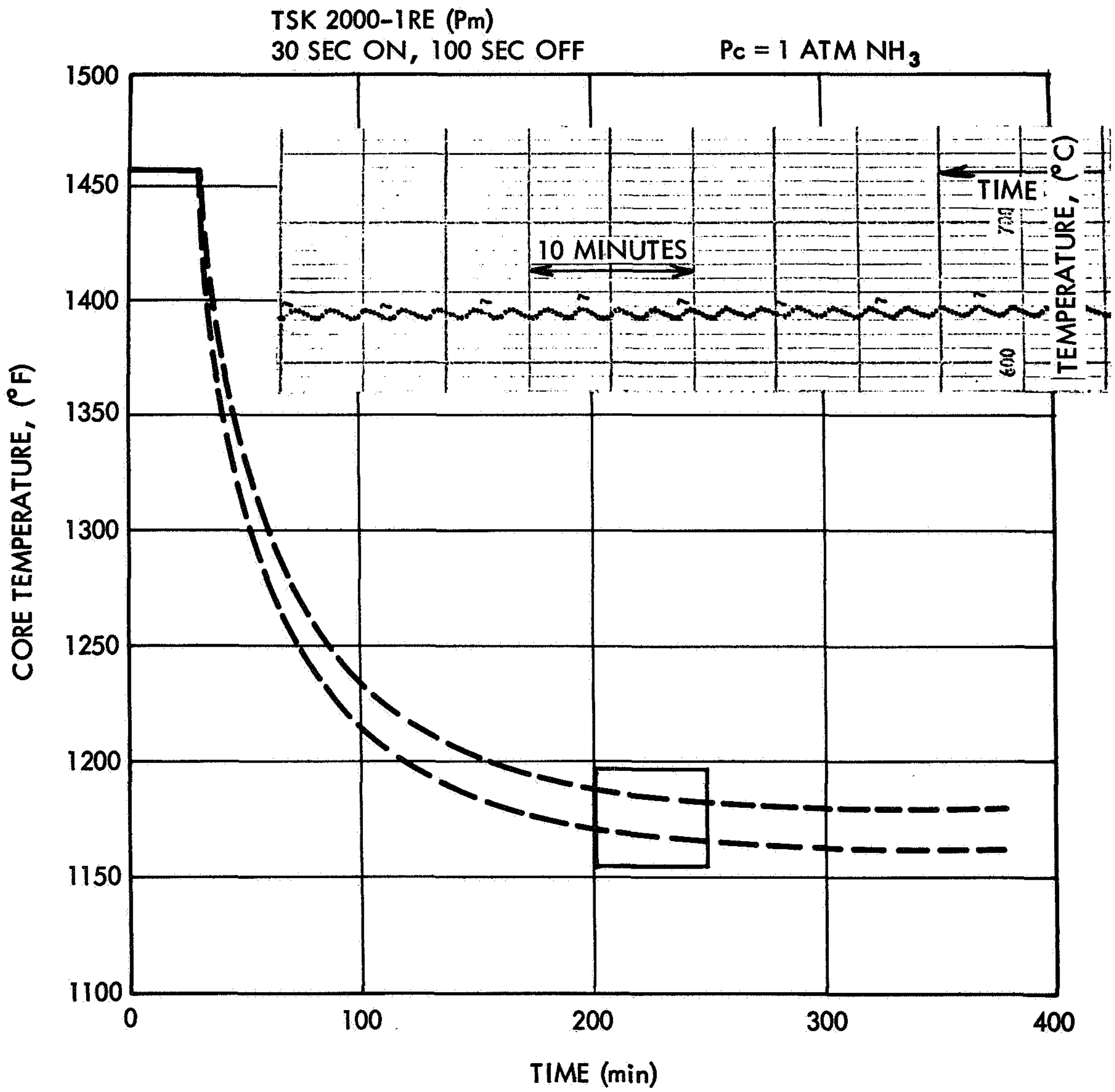

TSK 2000-1RE (Pm)
5 SEC ON, 50 SEC OFF
Pc = 1 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
250
TIME (min)



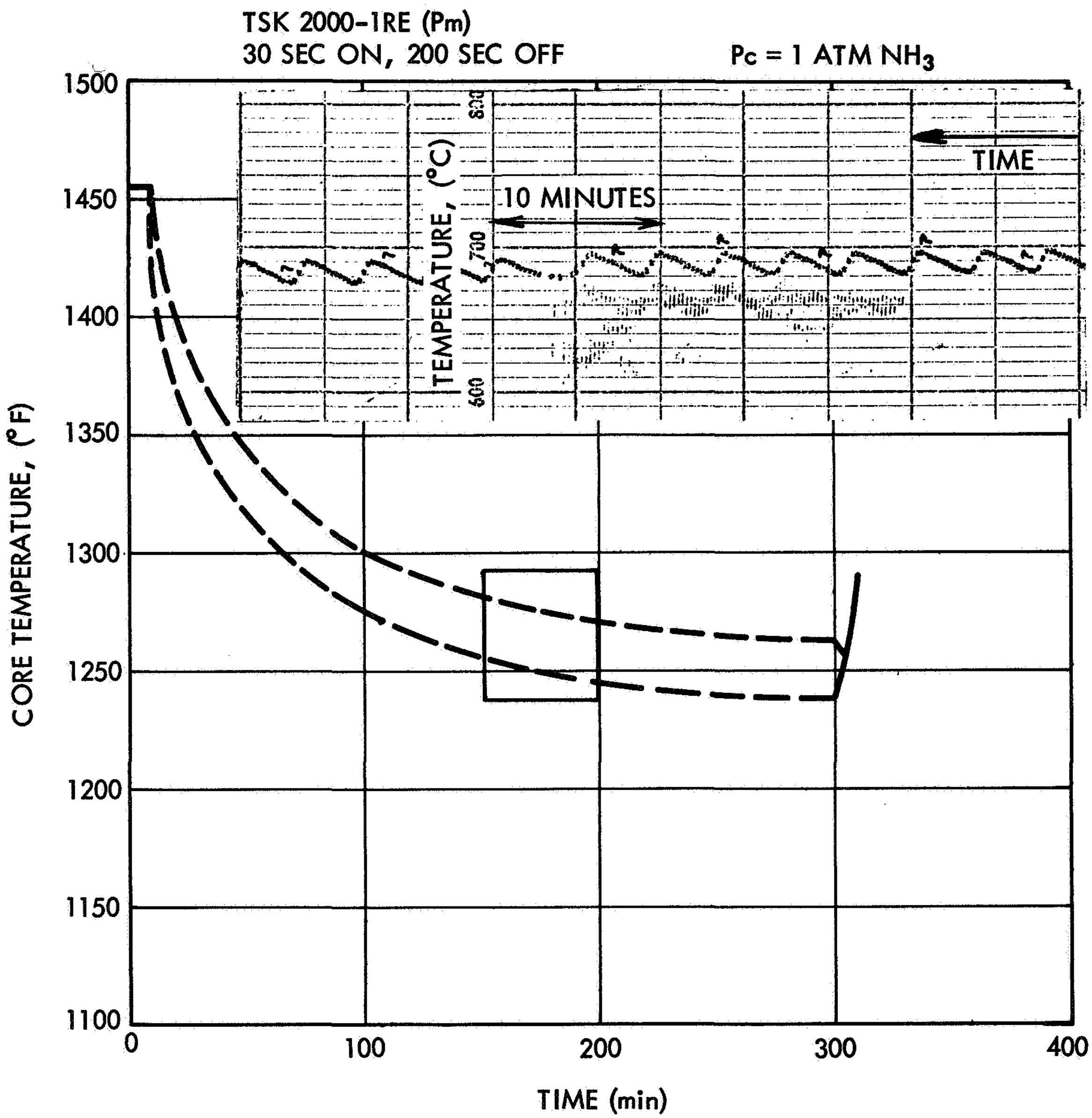
TSK 2000-1RE (Pm)
5 SEC ON, 100 SEC OFF
Pc = 1 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
TIME (min)
TEMPERATURE, (°C)
TIME
10 MINUTES

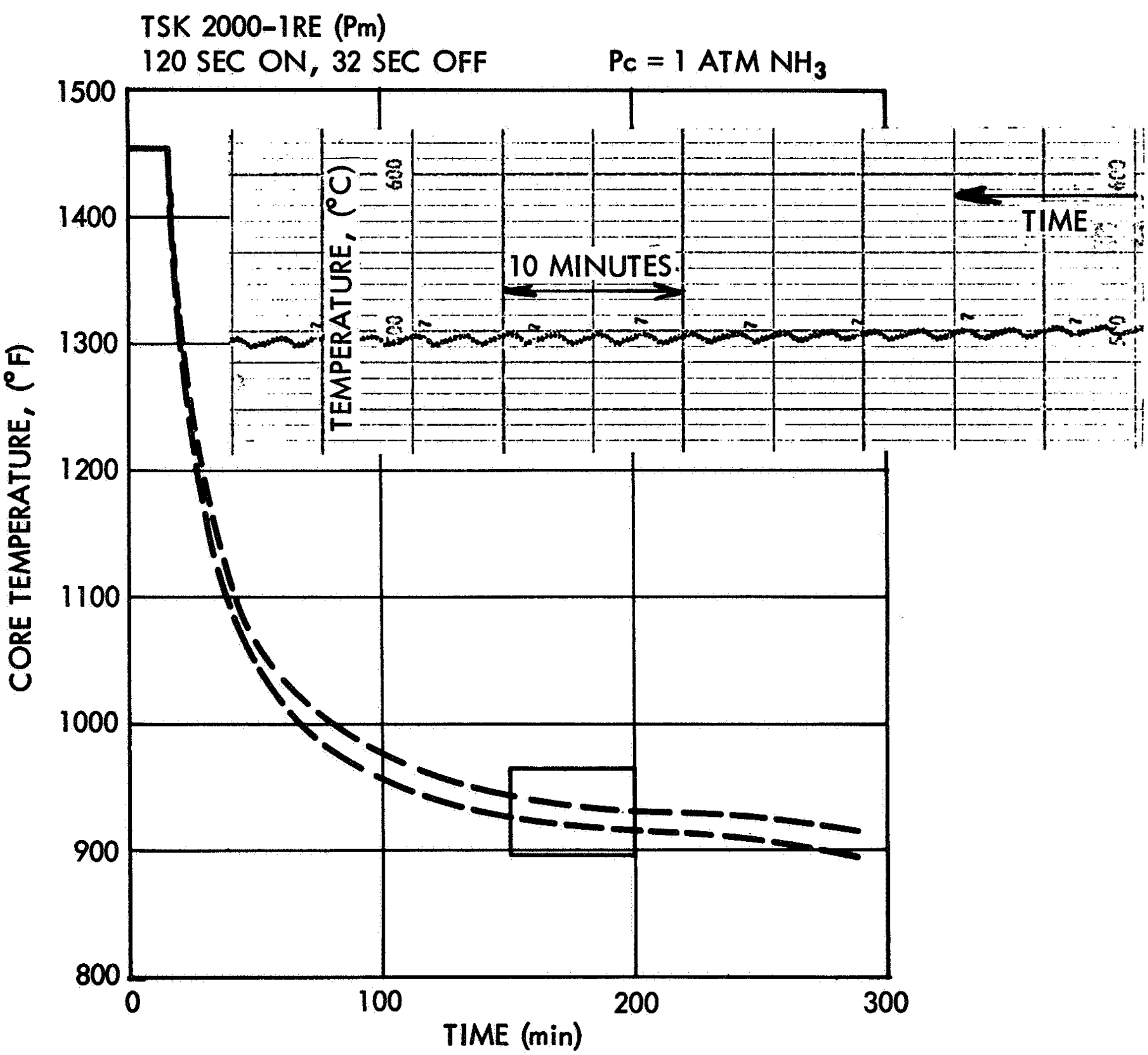

TSK 2000-1RE (Pm)
5 SEC ON, 200 SEC OFF
Pc = 1 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
250
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
900
800
700

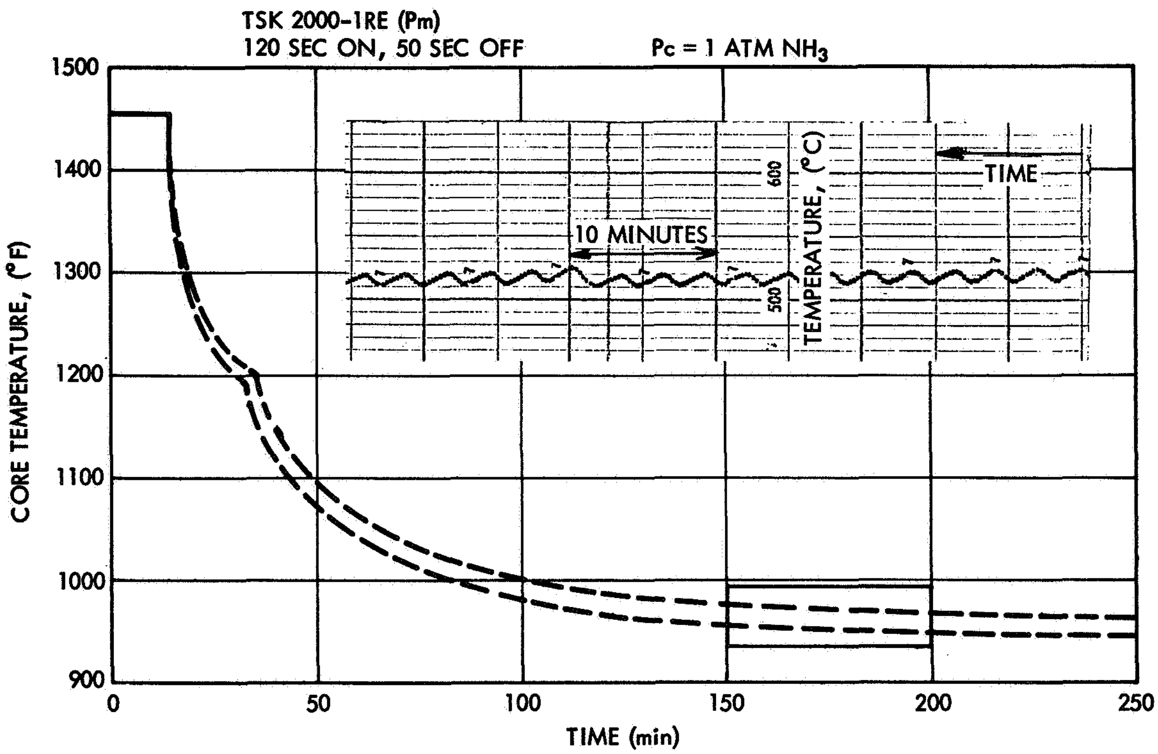

TSK 2000-1RE (pm)
30 SEC ON, 50 SEC OFF
Pc = 1 ATM NH_3
CORE TEMPERATURE, (°F)
TIME (min)
10 MINUTES
TEMPERATURE, (°C)
TIME
700
600
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200

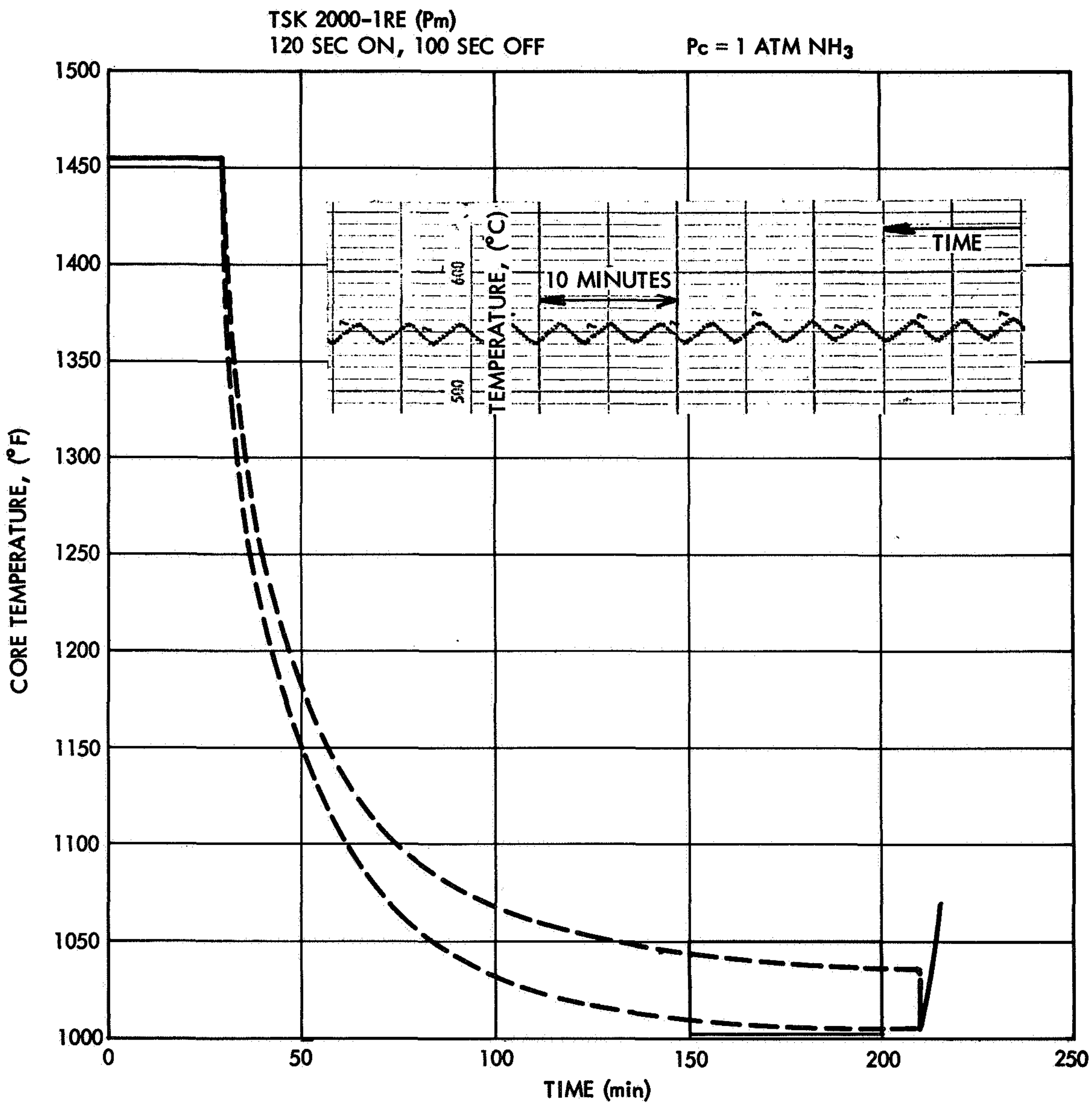


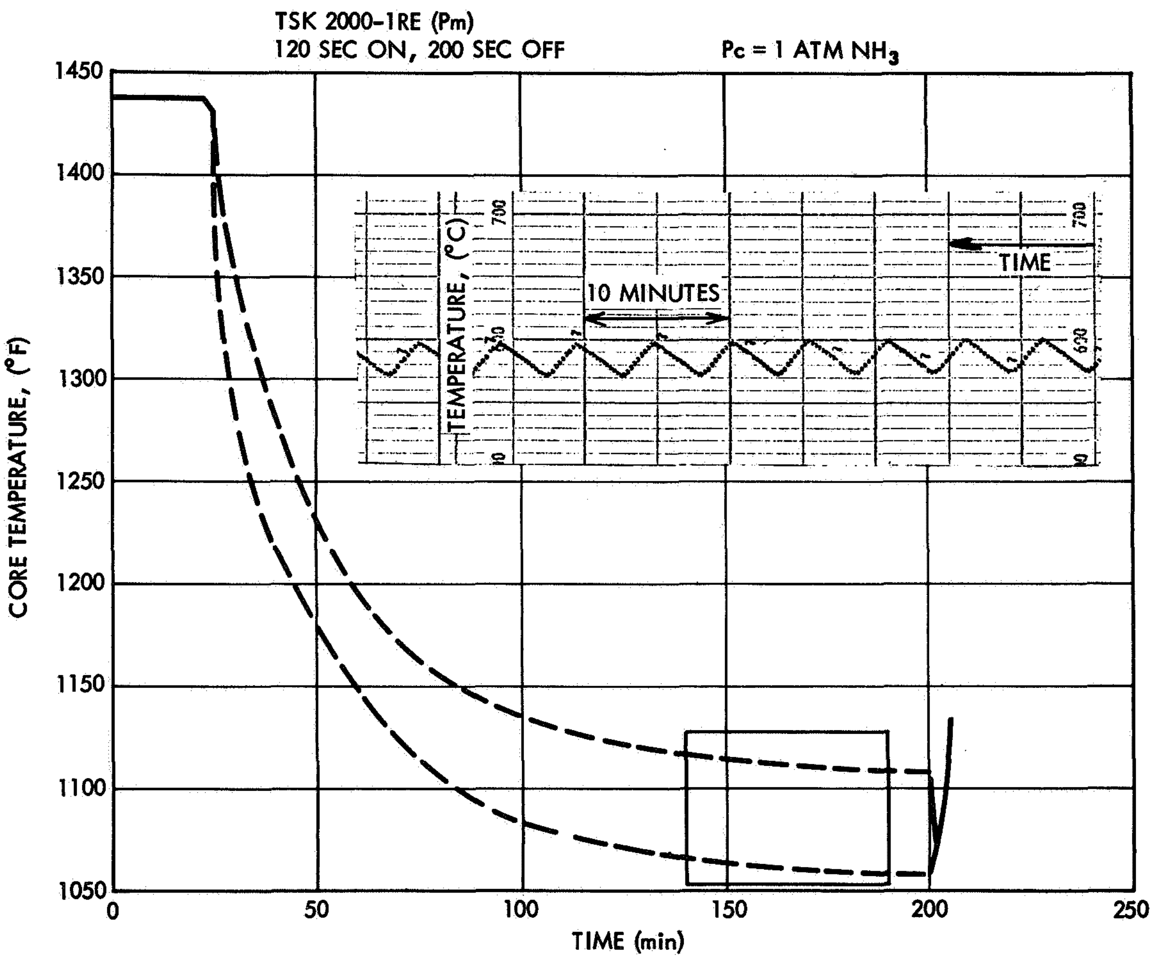
TSK 2000-1RE (Pm)
30 SEC ON, 100 SEC OFF
Pc = 1 ATM NH_3
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
100
200
300
400
CORE TEMPERATURE, (°F)
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
700
600


TSK 2000-1RE (Pm)
30 SEC ON, 200 SEC OFF
Pc = 1 ATM NH_3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
100
200
300
400
TIME (min)
TEMPERATURE, (°C)
800
700
600
10 MINUTES
TIME

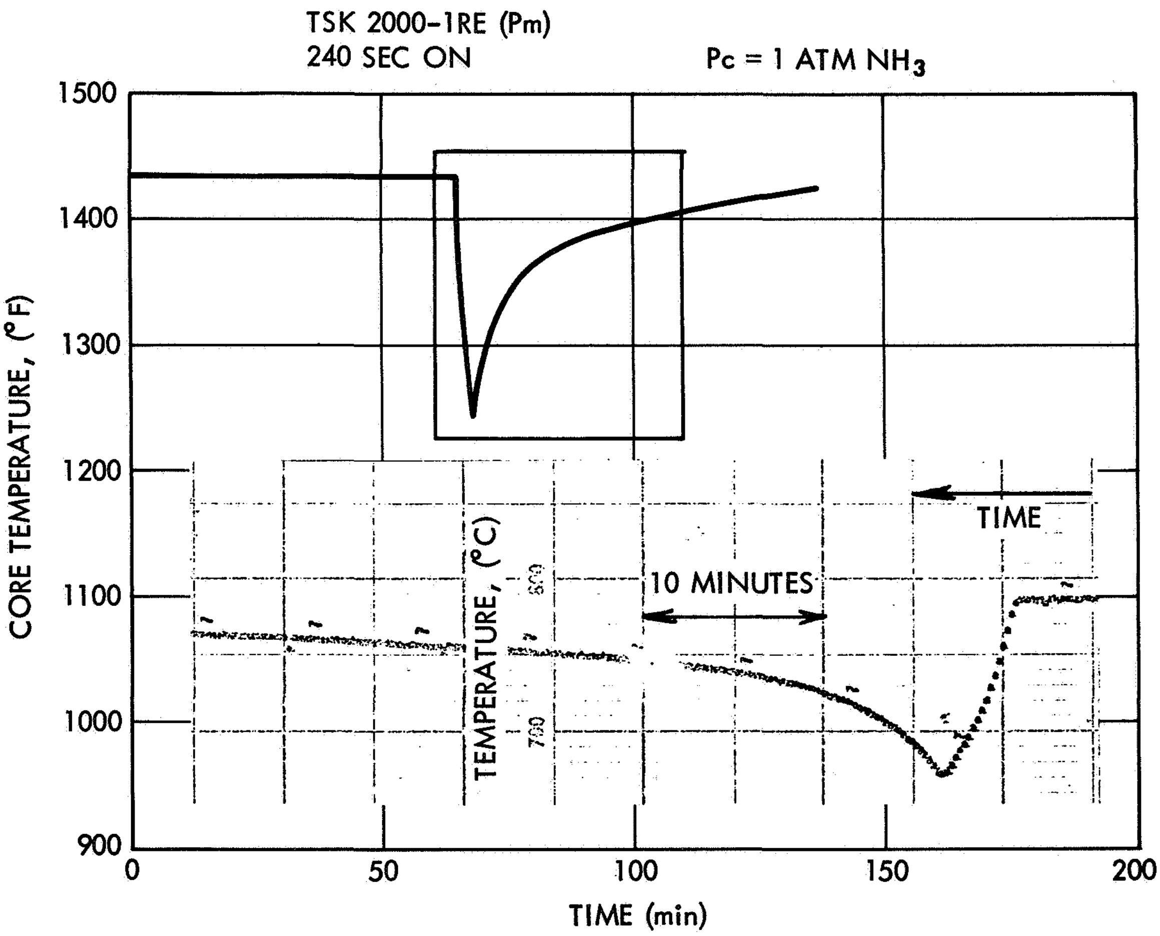

TSK 2000-1RE (Pm)
120 SEC ON, 32 SEC OFF
Pc = 1 ATM NH3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
800
0
100
200
300
TIME (min)
TEMPERATURE, (°C)
600
500
10 MINUTES
TIME

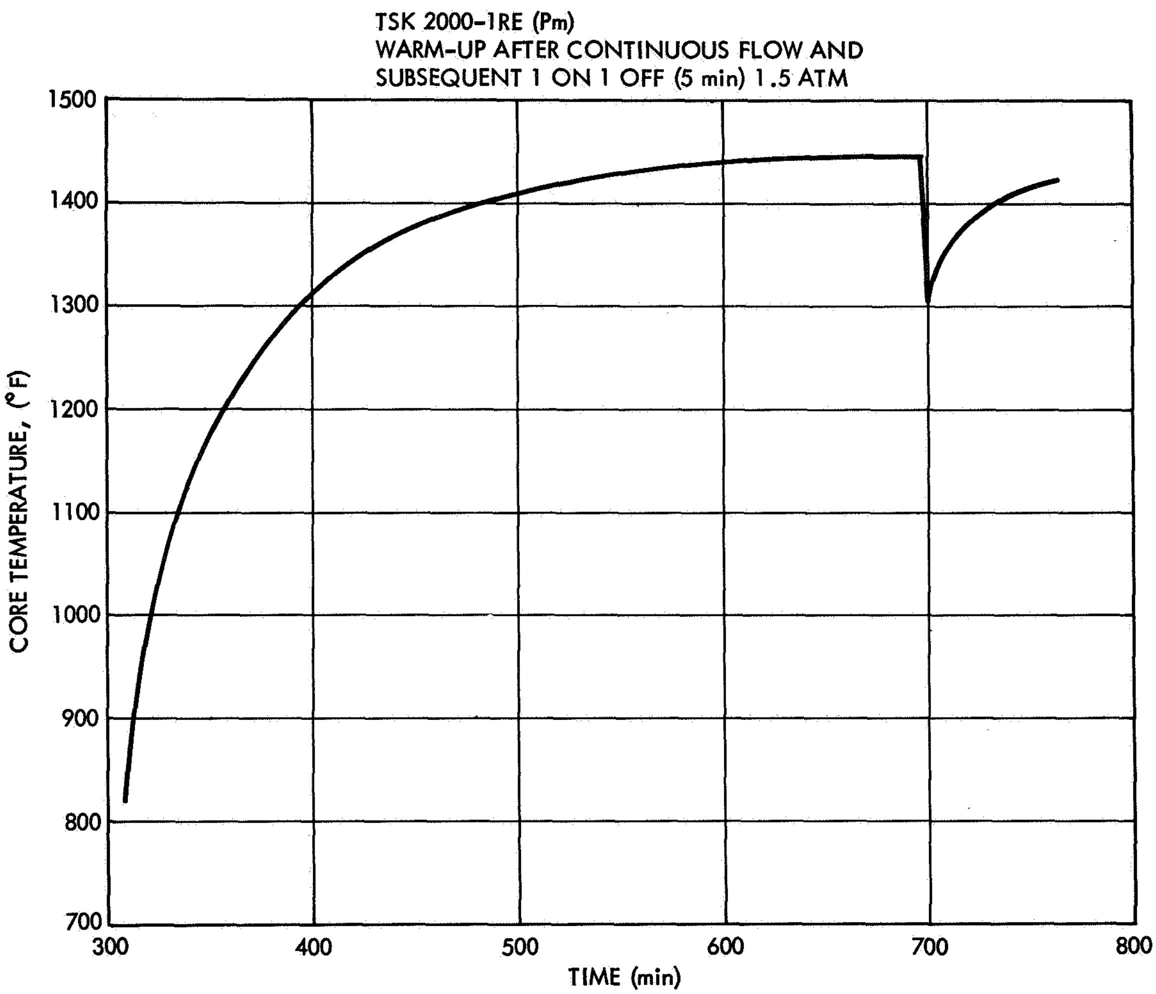

TSK 2000-1RE (Pm)
120 SEC ON, 50 SEC OFF
Pc = 1 ATM NH3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
0
50
100
150
200
250
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
600
500

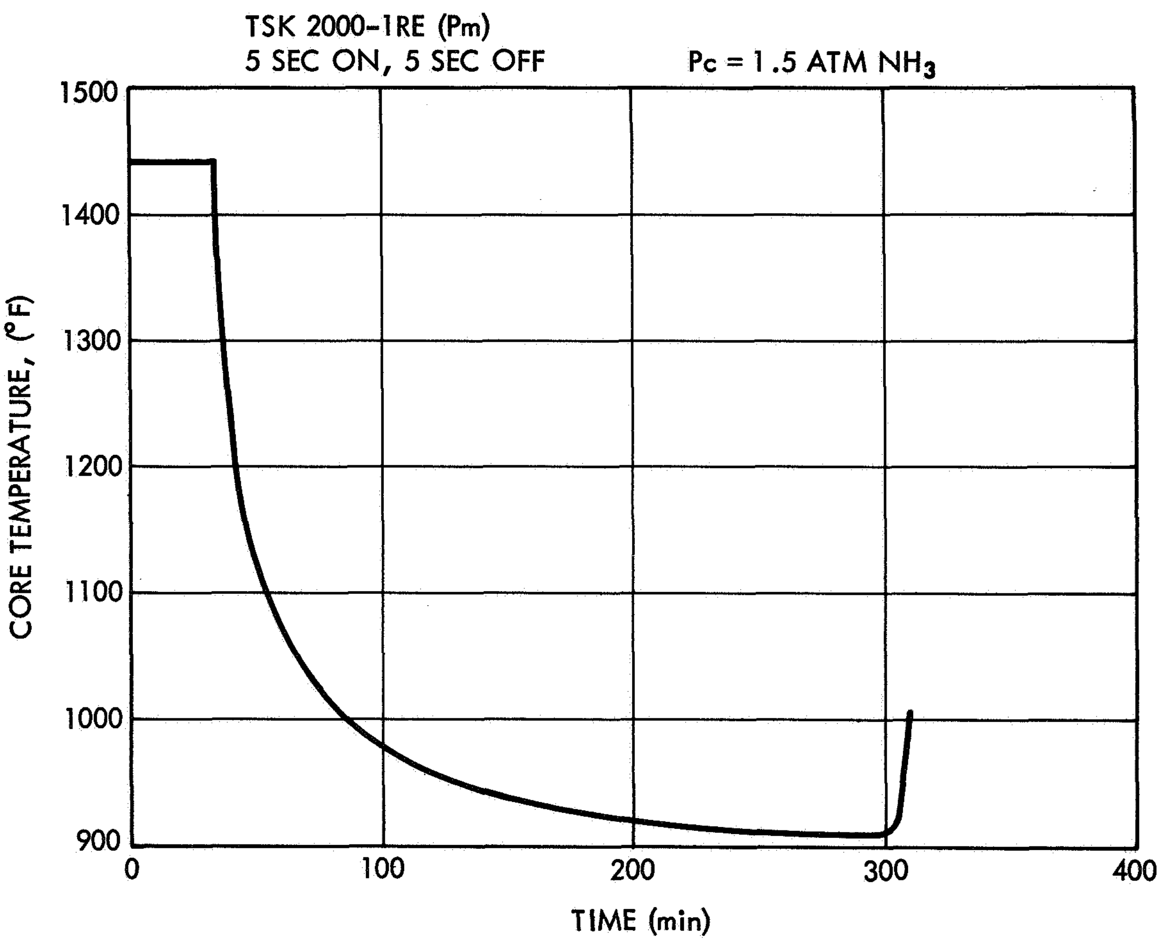

TSK 2000-1RE (Pm)
120 SEC ON, 100 SEC OFF
Pc = 1 ATM NH_3
CORE TEMPERATURE, (°F)
TIME (min)
1500
1450
1400
1350
1300
1250
1200
1150
1100
1050
1000
0
50
100
150
200
250
TEMPERATURE, (°C)
600
500
10 MINUTES
TIME

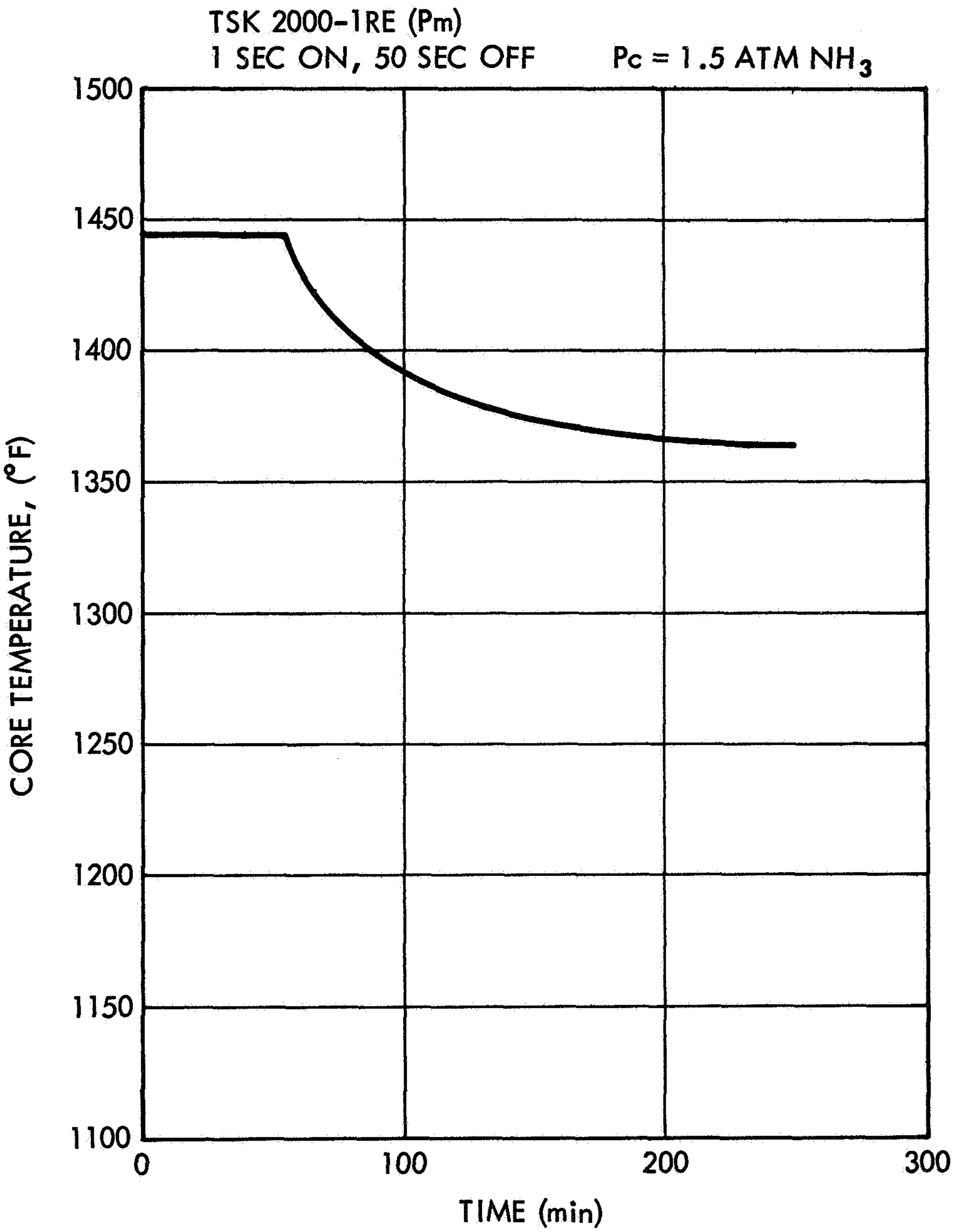


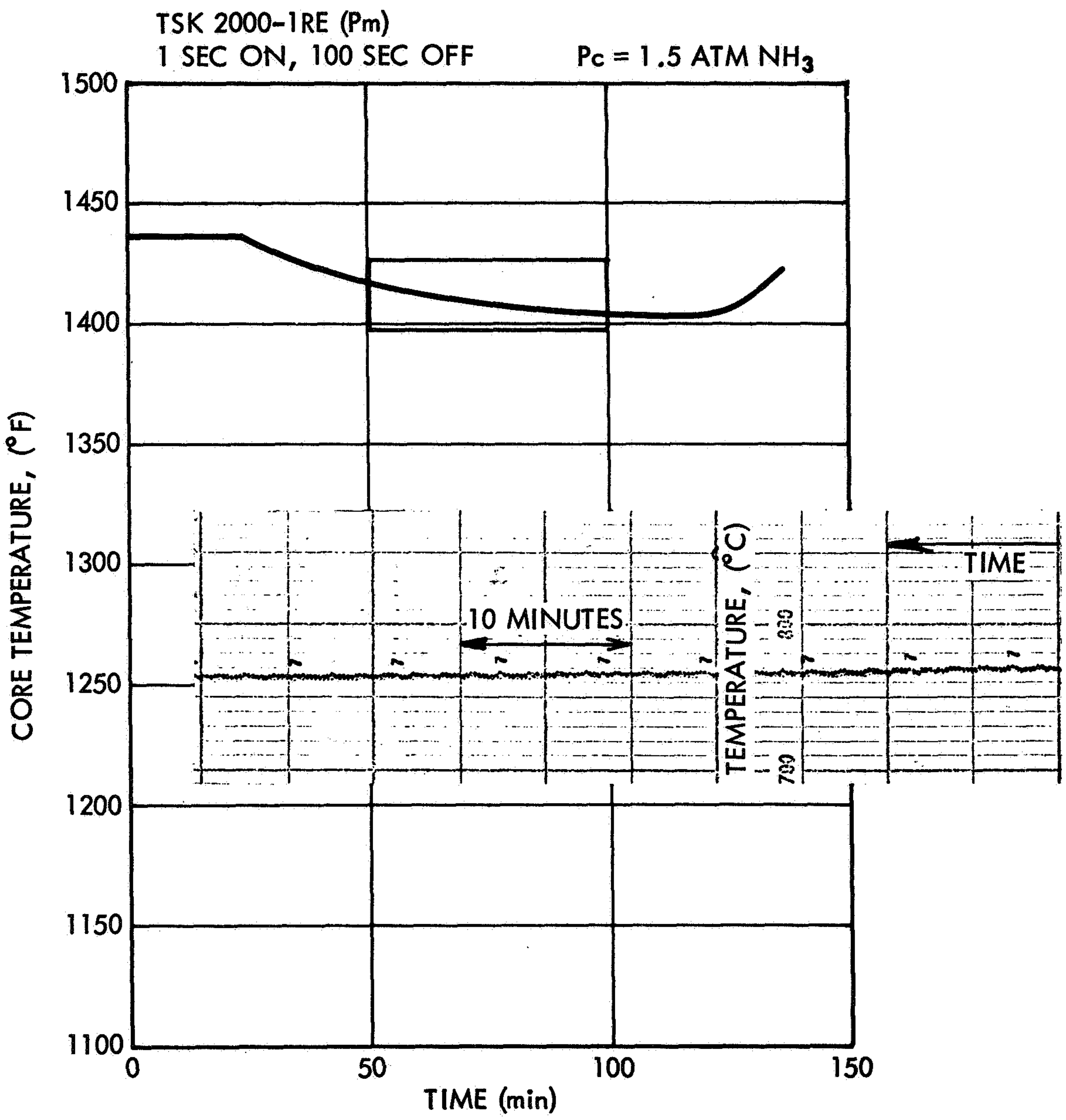
TSK 2000-1RE (Pm)
120 SEC ON, 200 SEC OFF
Pc = 1 ATM NH_3
CORE TEMPERATURE, (°F)
TIME (min)
TEMPERATURE, (°C)
10 MINUTES
TIME
1450
1400
1350
1300
1250
1200
1150
1100
1050
0
50
100
150
200
250

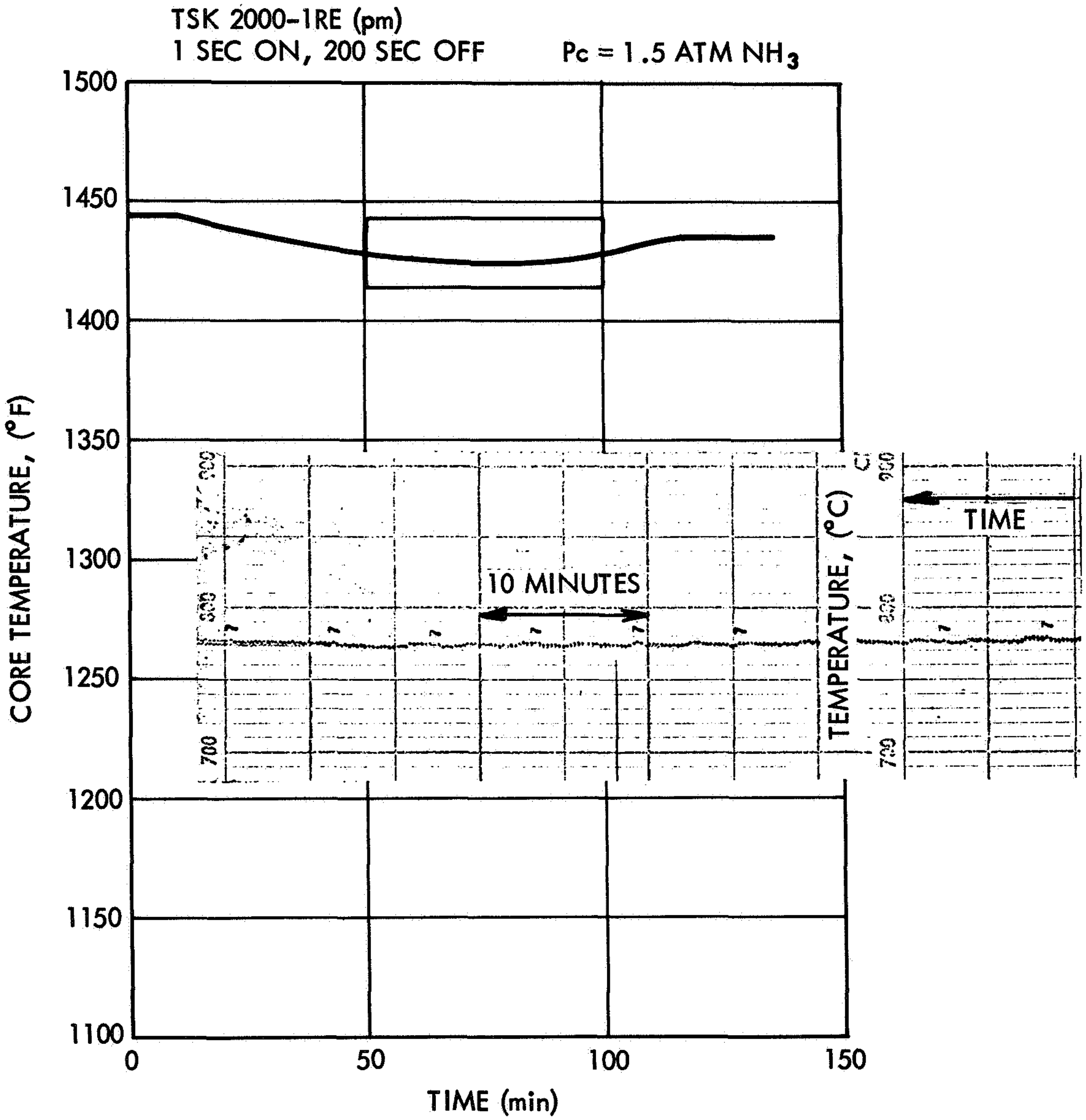

TSK 2000-1RE (Pm)
240 SEC ON
Pc = 1 ATM NH3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
TEMPERATURE, (°C)
10 MINUTES
TIME
0
50
100
150
200
TIME (min)

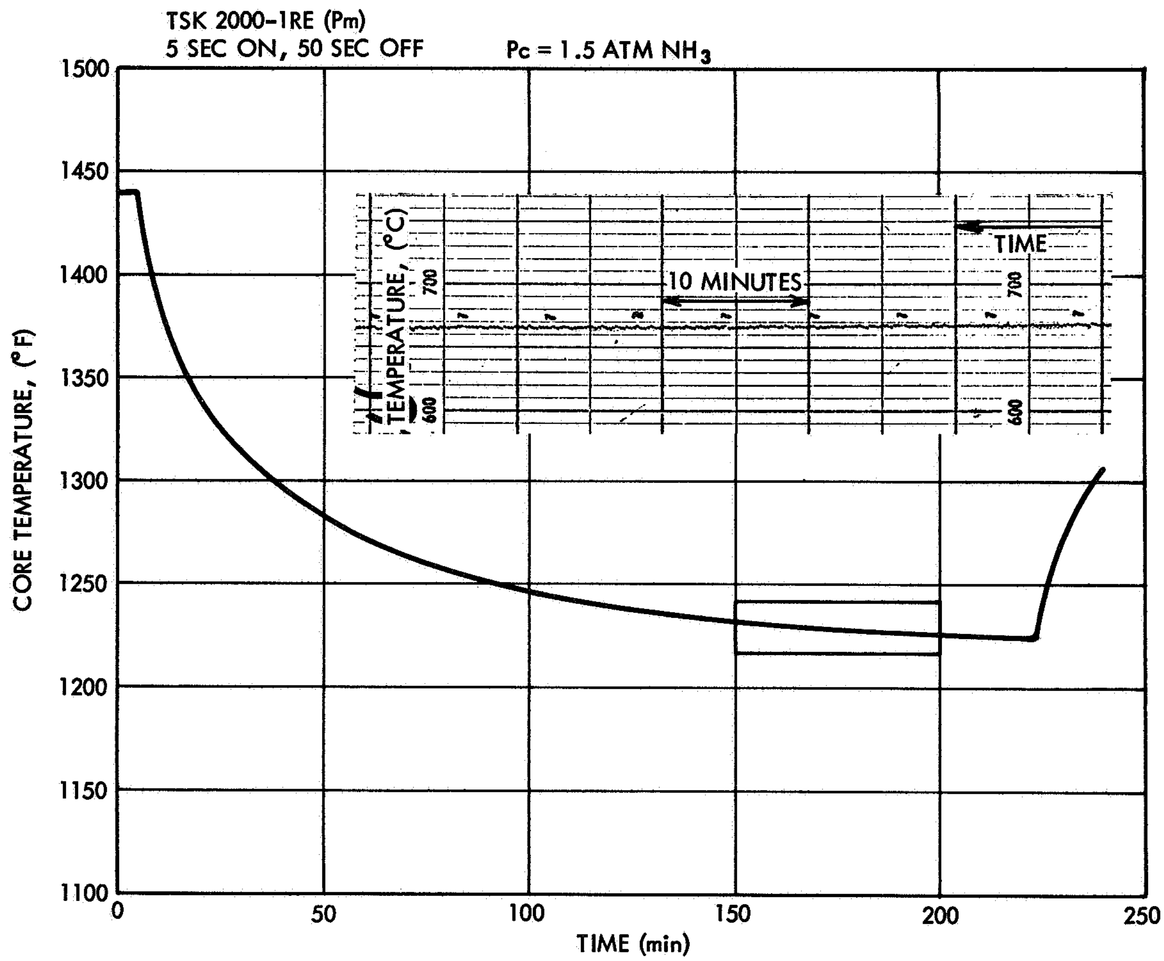

TSK 2000-1RE (Pm)
WARM-UP AFTER CONTINUOUS FLOW AND
SUBSEQUENT 1 ON 1 OFF (5 min) 1.5 ATM
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
800
700
300
400
500
600
700
800
TIME (min)


TSK 2000-1RE (Pm)
5 SEC ON, 5 SEC OFF
Pc = 1.5 ATM NH3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
0
100
200
300
400
TIME (min)

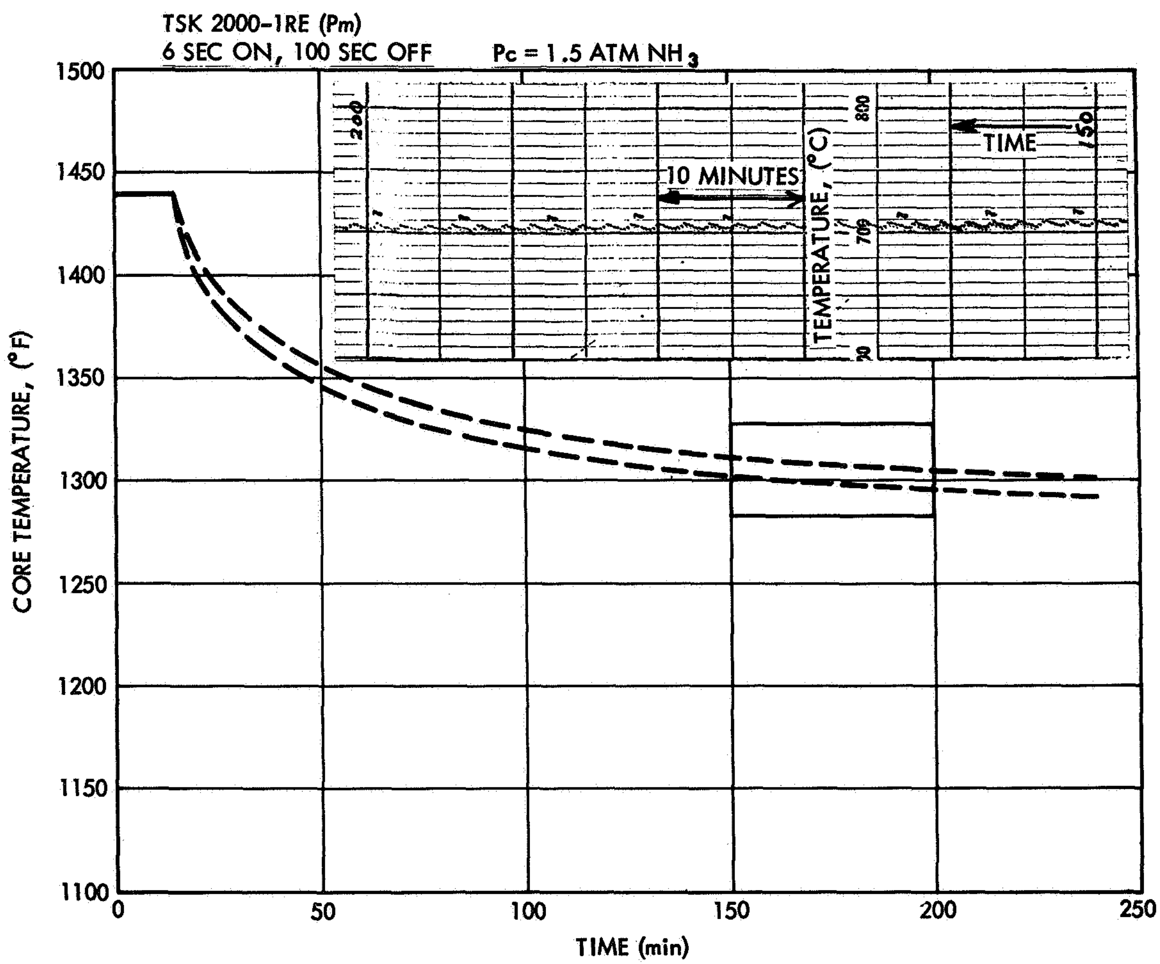

TSK 2000-1RE (Pm)
1 SEC ON, 50 SEC OFF
Pc = 1.5 ATM NH_3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
100
200
300
TIME (min)


TSK 2000-1RE (Pm)
1 SEC ON, 100 SEC OFF
Pc = 1.5 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
TIME (min)
10 MINUTES
TEMPERATURE, (°C)
TIME

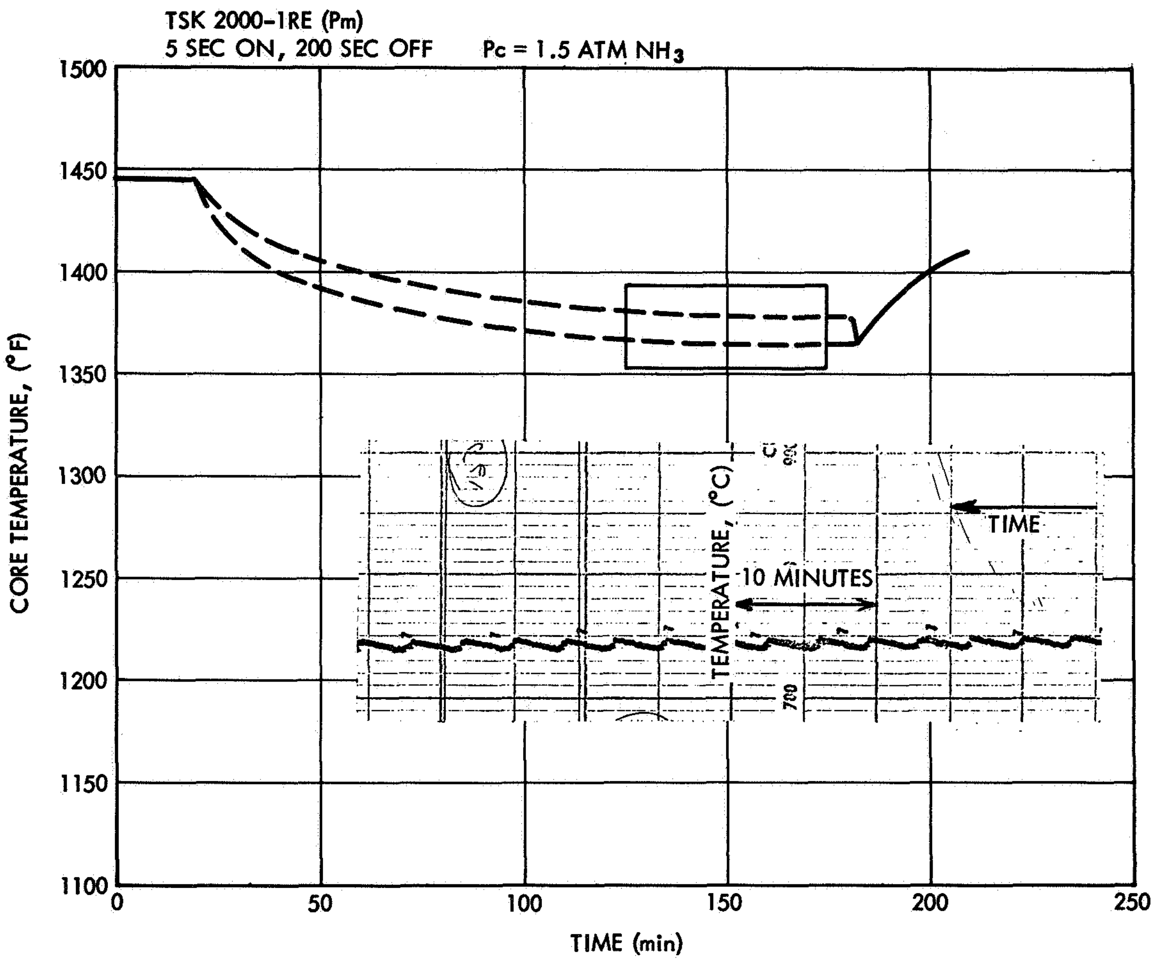

TSK 2000-1RE (pm)
1 SEC ON, 200 SEC OFF
Pc = 1.5 ATM NH_3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
TIME (min)
10 MINUTES
TEMPERATURE, (°C)
TIME
900
800
700



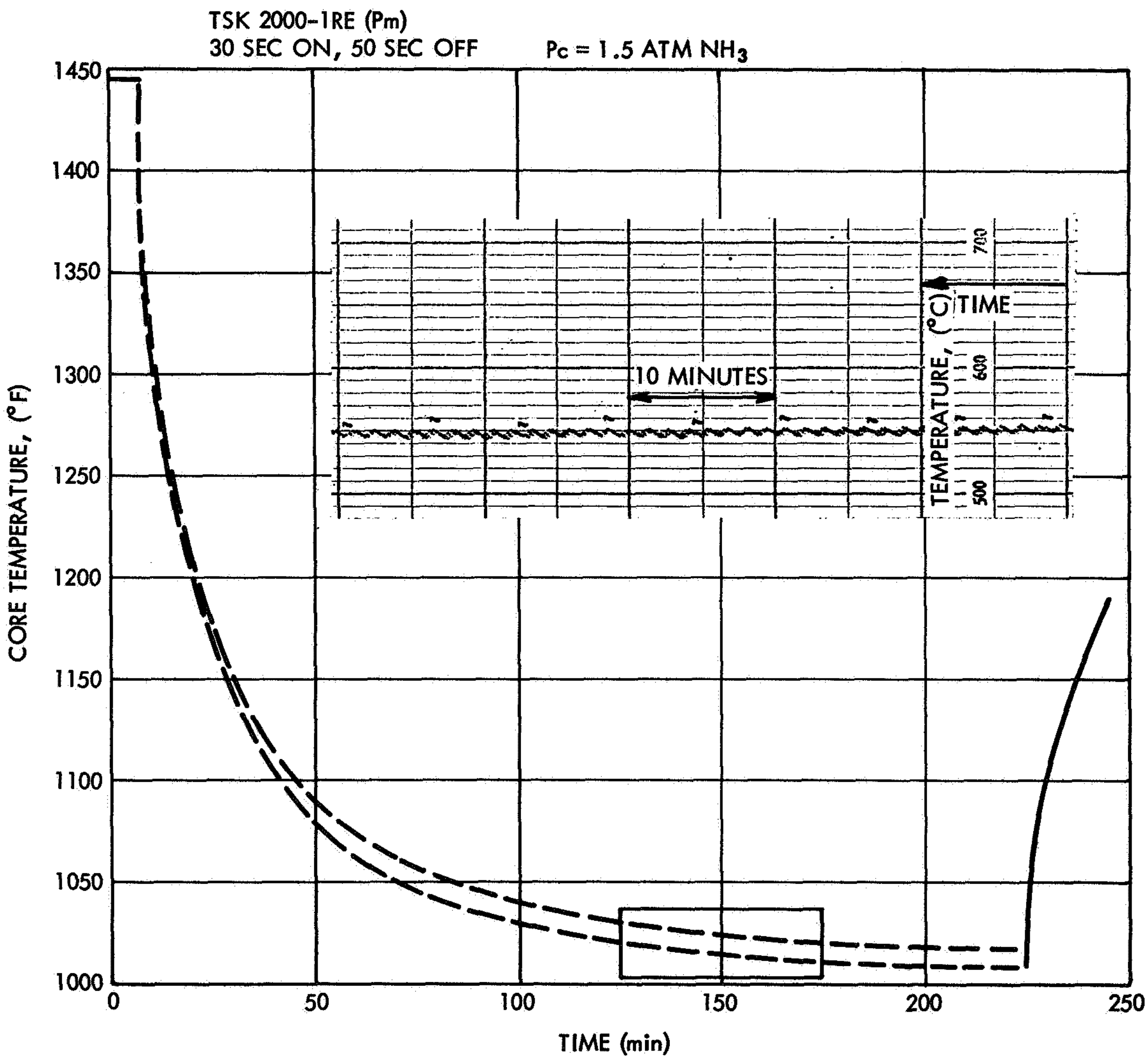
TSK 2000-1RE (Pm)
5 SEC ON, 50 SEC OFF
Pc = 1.5 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
250
TIME (min)
TEMPERATURE, (°C)
700
600
10 MINUTES
TIME

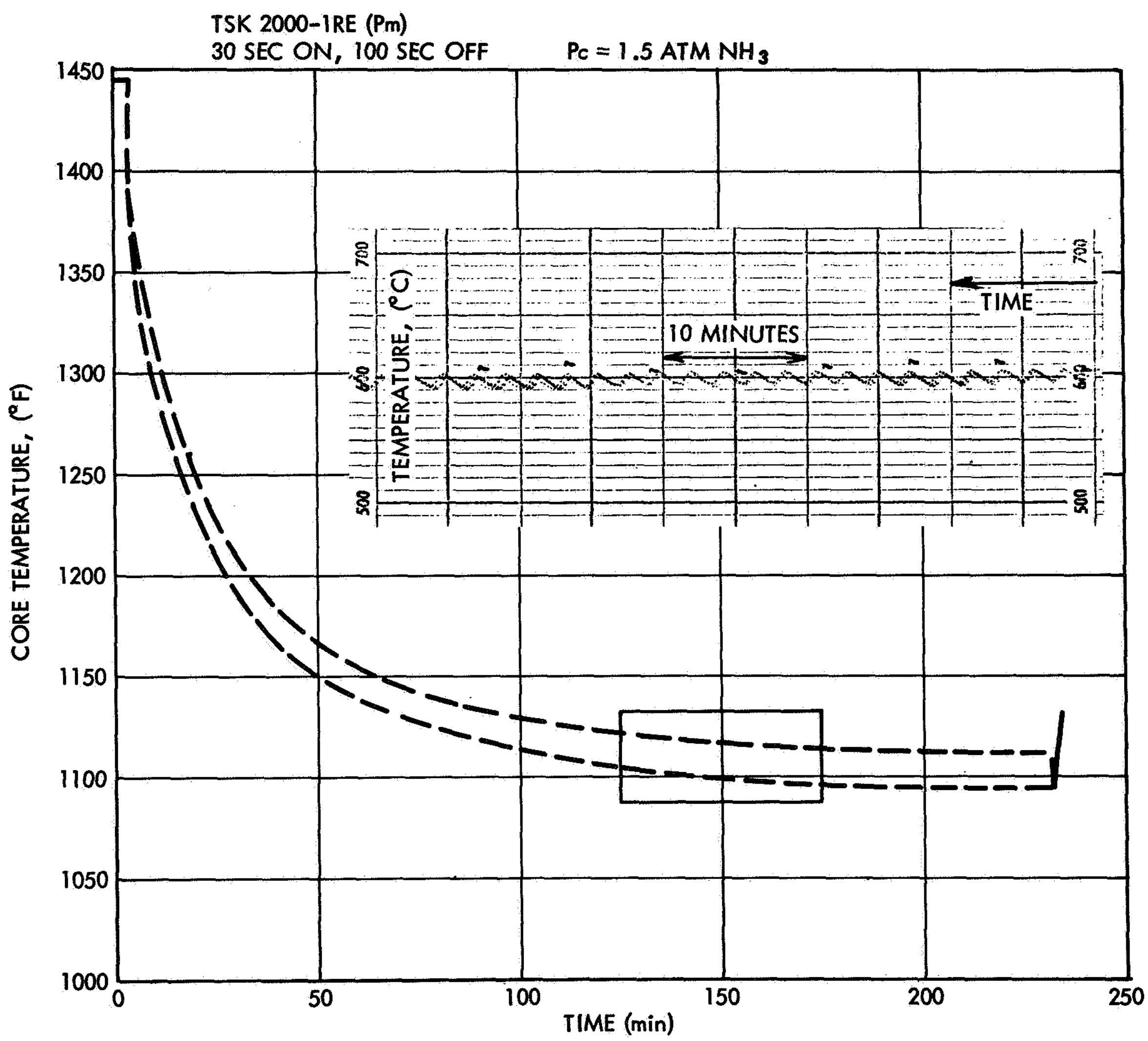


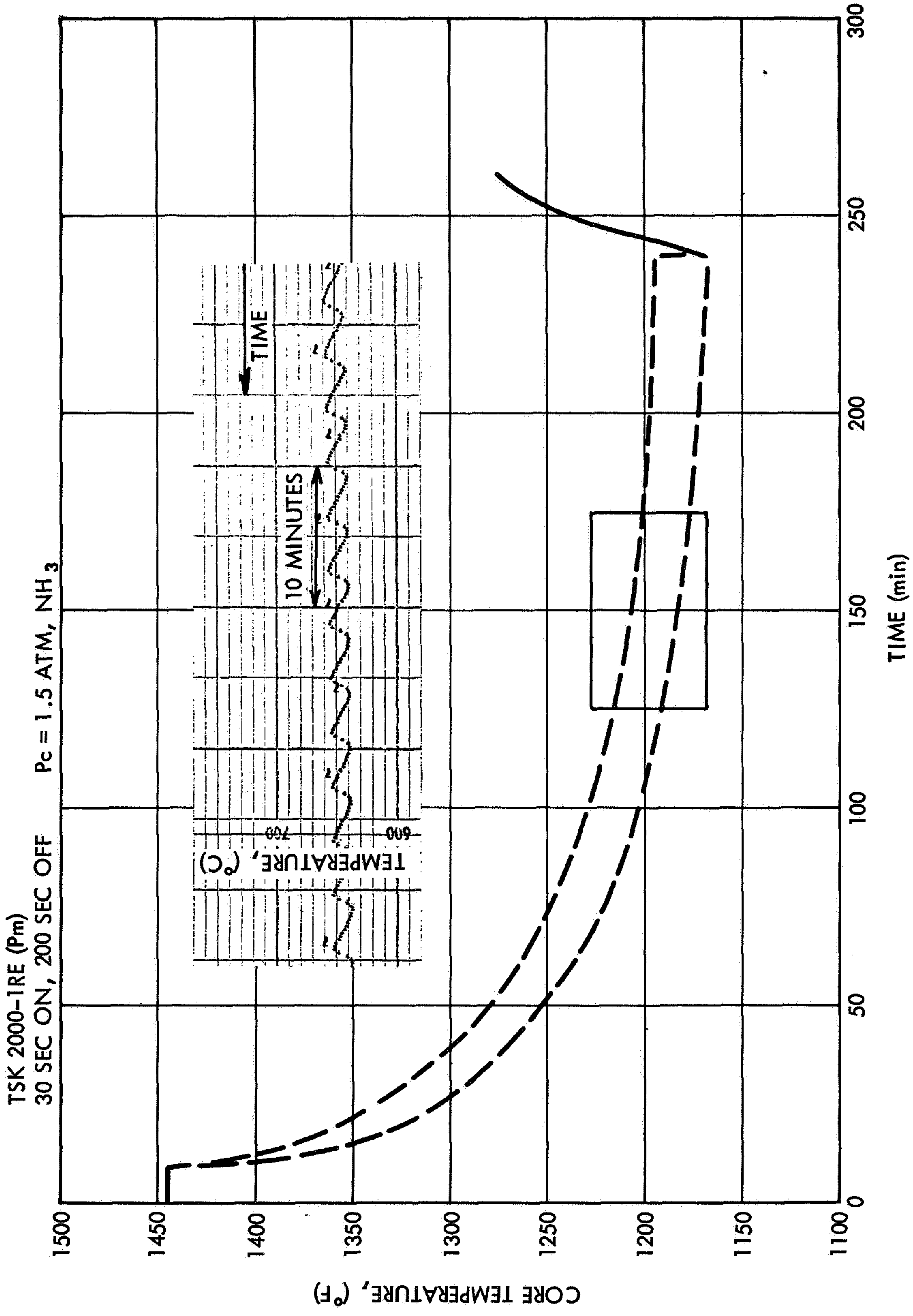
TSK 2000-1RE (Pm)
6 SEC ON, 100 SEC OFF
Pc = 1.5 ATM NH3
10 MINUTES
TIME
TEMPERATURE, (°C)
CORE TEMPERATURE, (°F)
TIME (min)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
250

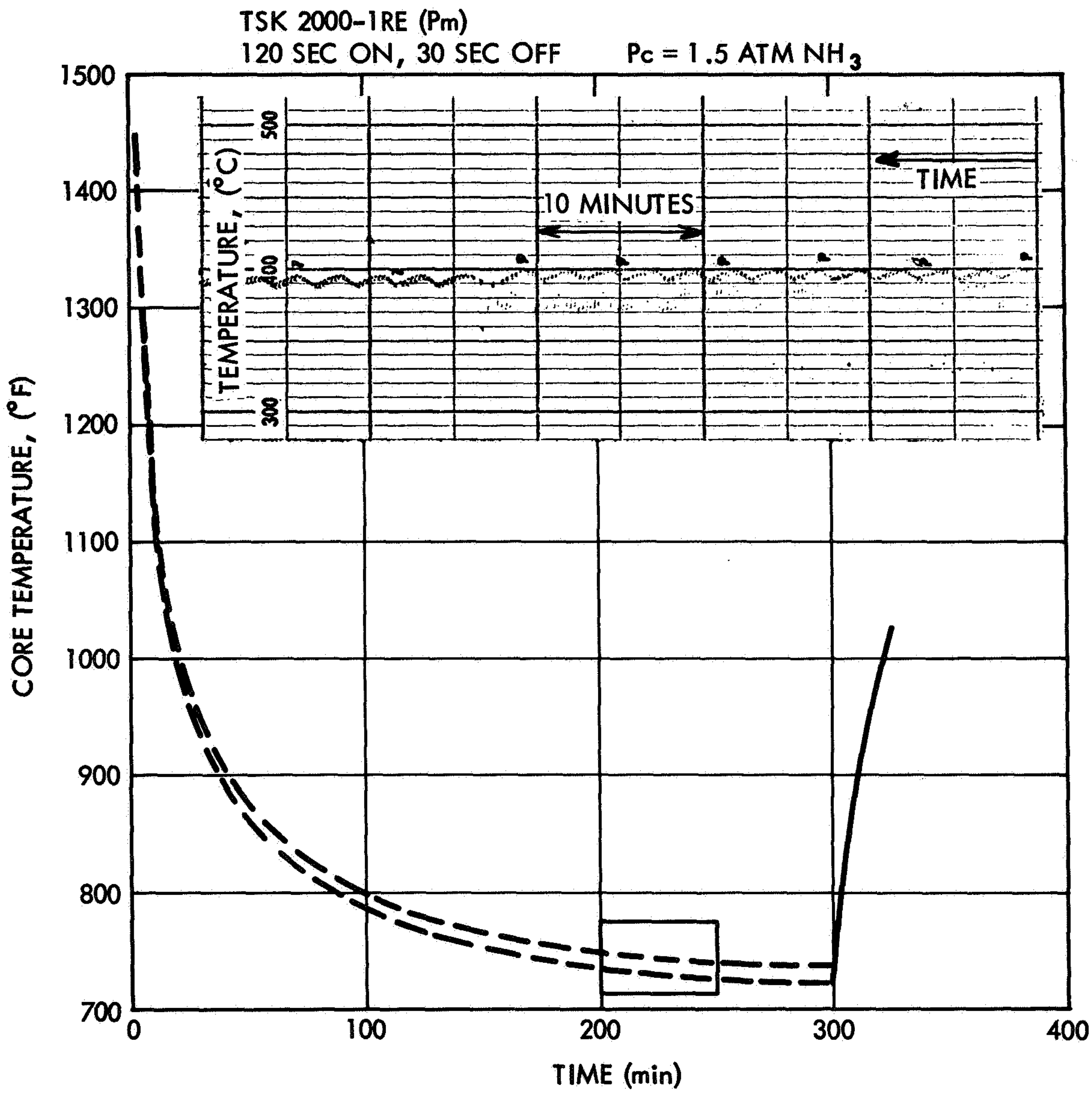


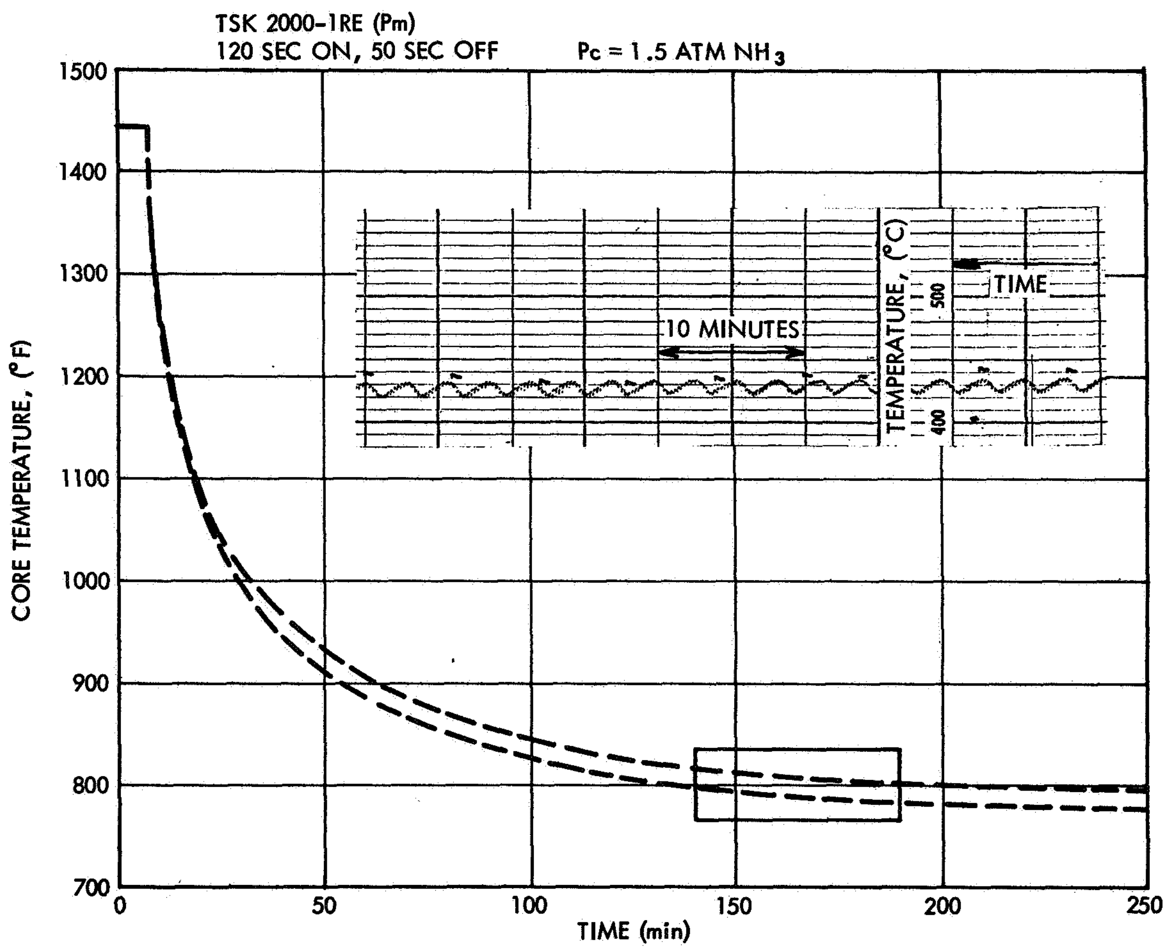
TSK 2000-1RE (Pm)
5 SEC ON, 200 SEC OFF
Pc = 1.5 ATM NH3
1500
1450
1400
1350
1300
1250
1200
1150
1100
CORE TEMPERATURE, (°F)
0
50
100
150
200
250
TIME (min)
TEMPERATURE, (°C)
TIME
10 MINUTES
700

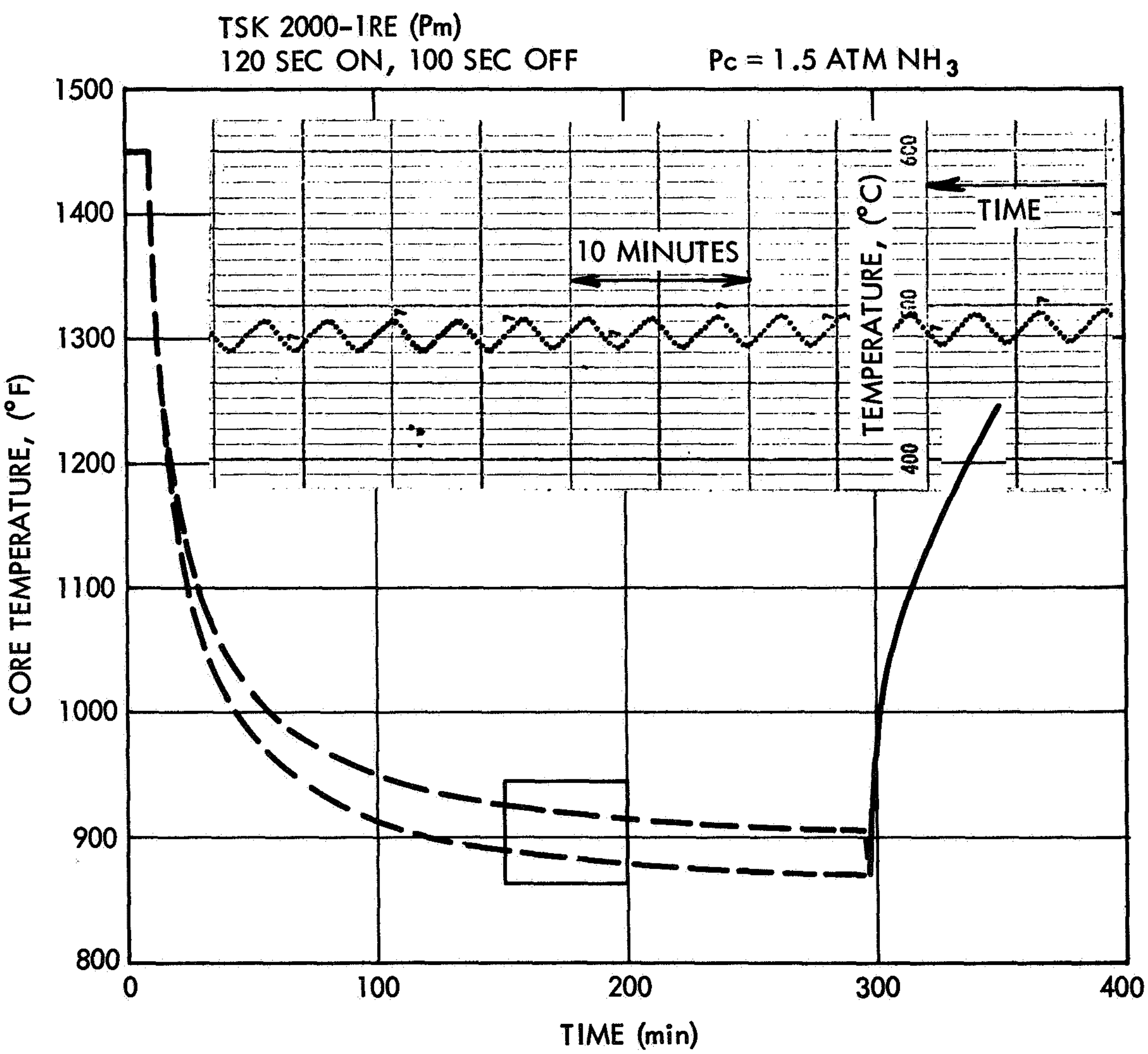

TSK 2000-1RE (Pm)
30 SEC ON, 50 SEC OFF
Pc = 1.5 ATM NH3
CORE TEMPERATURE, (°F)
1450
1400
1350
1300
1250
1200
1150
1100
1050
1000
0
50
100
150
200
250
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
700
600
500


TSK 2000-1RE (Pm)
30 SEC ON, 100 SEC OFF
Pc = 1.5 ATM NH3
CORE TEMPERATURE, (°F)
TIME (min)
TEMPERATURE, (°C)
10 MINUTES
TIME
1450
1400
1350
1300
1250
1200
1150
1100
1050
1000
0
50
100
150
200
250
700
500

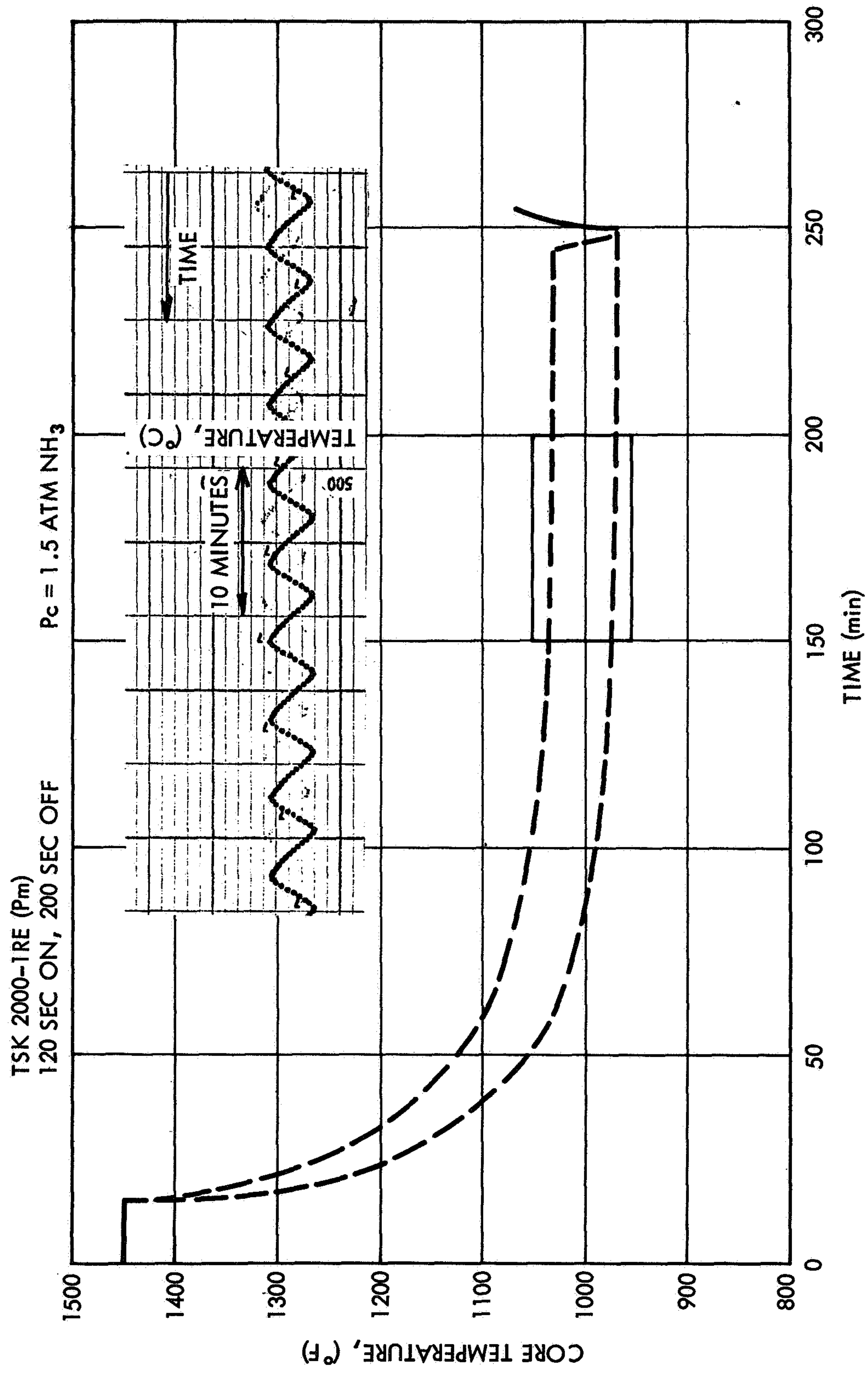

TSK 2000-1RE (Pm)
30 SEC ON, 200 SEC OFF
Pc = 1.5 ATM, NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
TIME (min)
0
50
100
150
200
250
300
TEMPERATURE, (°C)
700
600
10 MINUTES
TIME

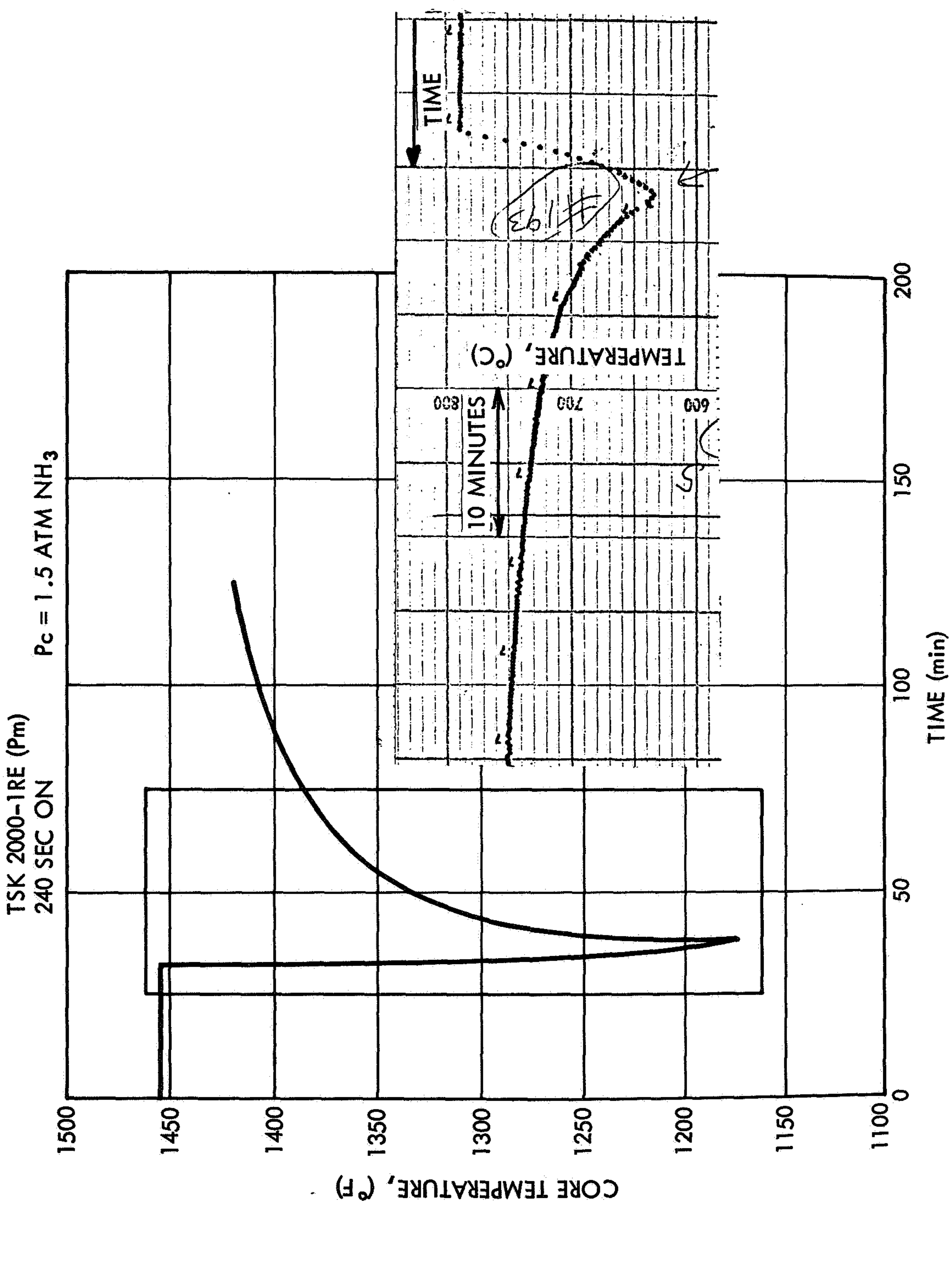

TSK 2000-1RE (Pm)
120 SEC ON, 30 SEC OFF
Pc = 1.5 ATM NH_3
1500
1400
1300
1200
1100
1000
900
800
700
CORE TEMPERATURE, (°F)
0
100
200
300
400
TIME (min)
TEMPERATURE, (°C)
500
400
300
10 MINUTES
TIME


TSK 2000-1RE (Pm)
120 SEC ON, 50 SEC OFF
Pc = 1.5 ATM NH3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
800
700
0
50
100
150
200
250
TIME (min)
10 MINUTES
TEMPERATURE, (°C)
500
400
TIME

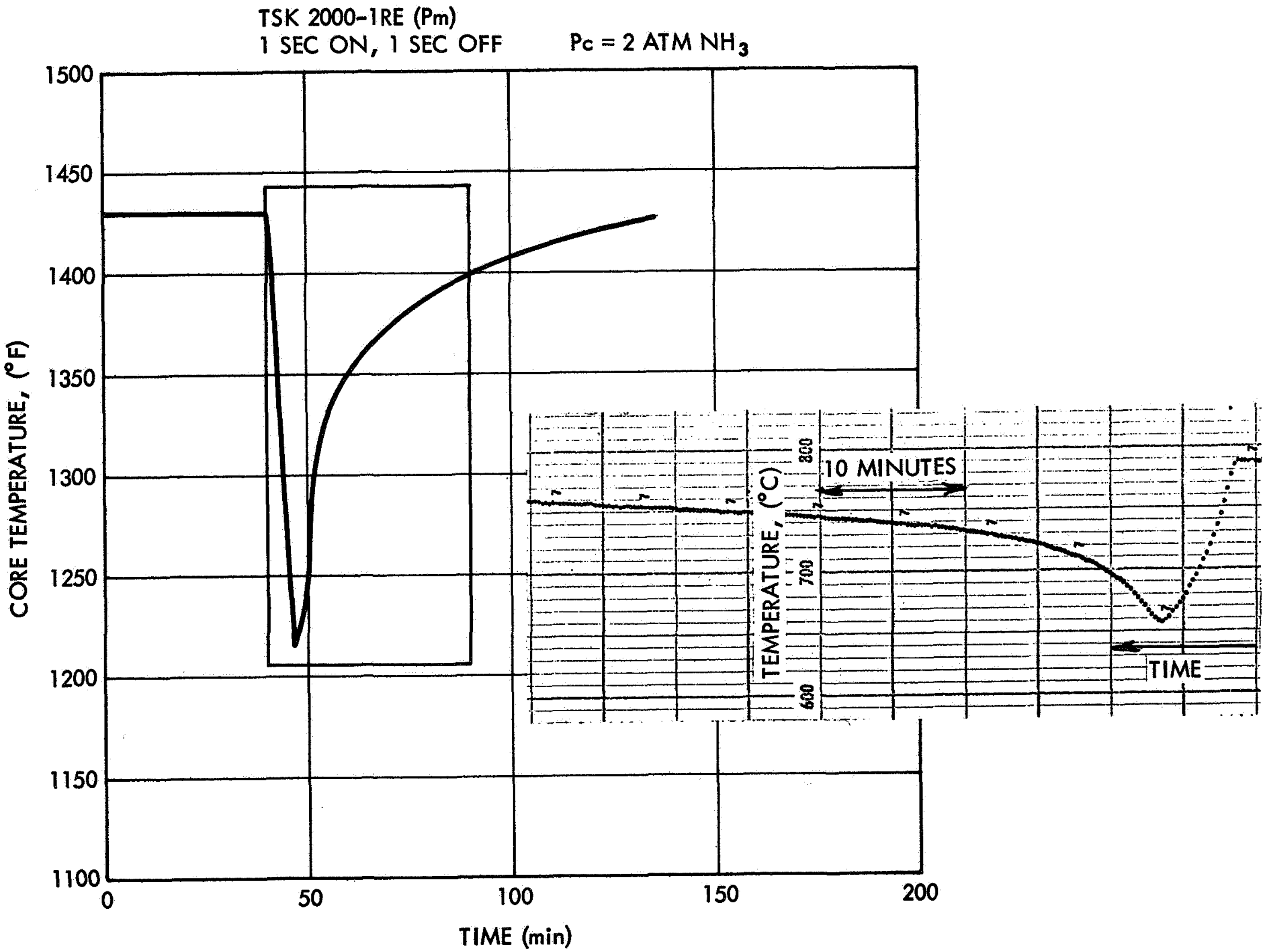


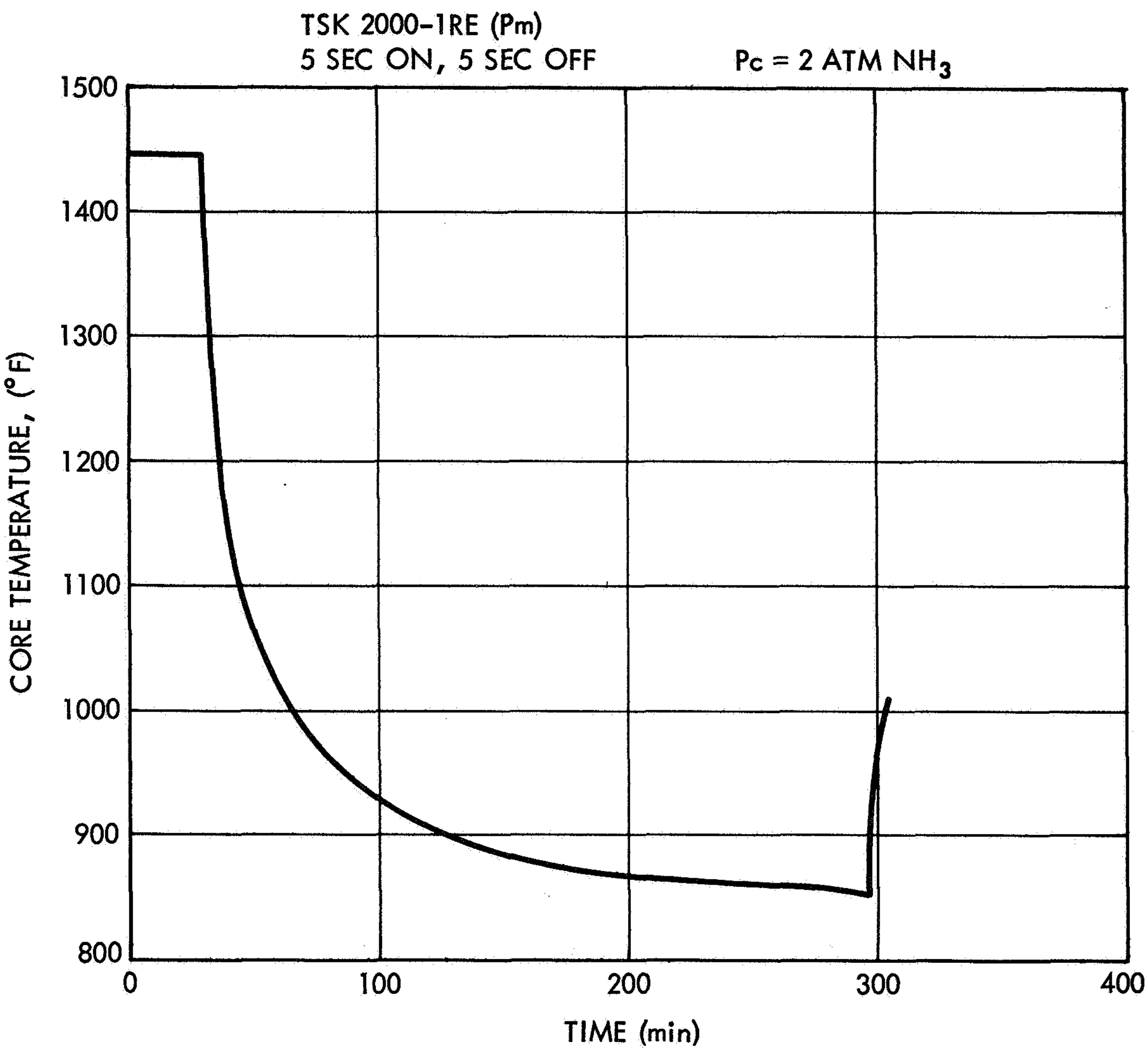
TSK 2000-1RE (Pm)
120 SEC ON, 100 SEC OFF
Pc = 1.5 ATM NH3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
800
0
100
200
300
400
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
600
500
400

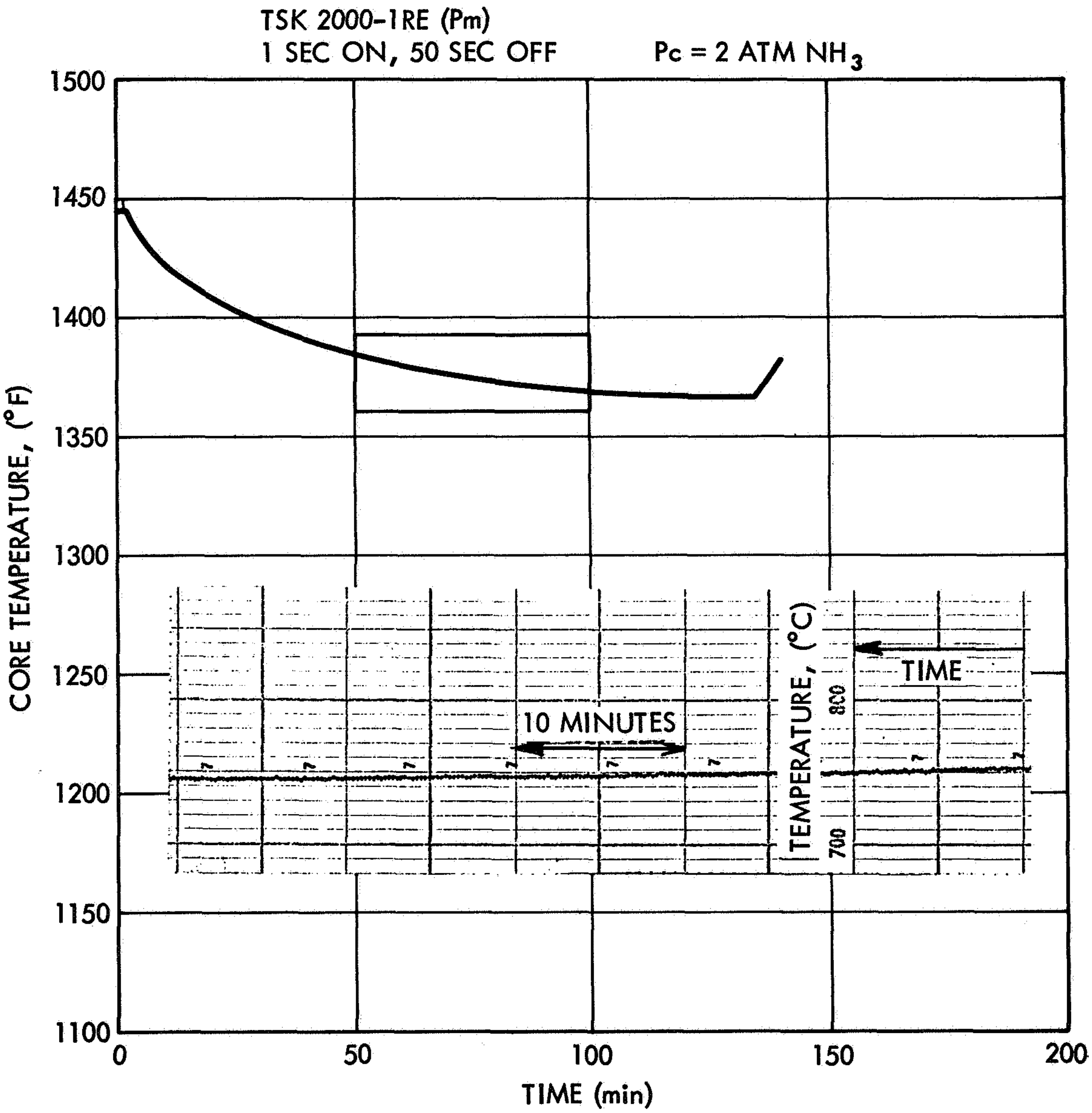

TSK 2000-1RE (Pm)
120 SEC ON, 200 SEC OFF
Pc = 1.5 ATM NH3
TIME
TEMPERATURE, (°C)
10 MINUTES
500
CORE TEMPERATURE, (°F)
TIME (min)
1500
1400
1300
1200
1100
1000
900
800
0
50
100
150
200
250
300



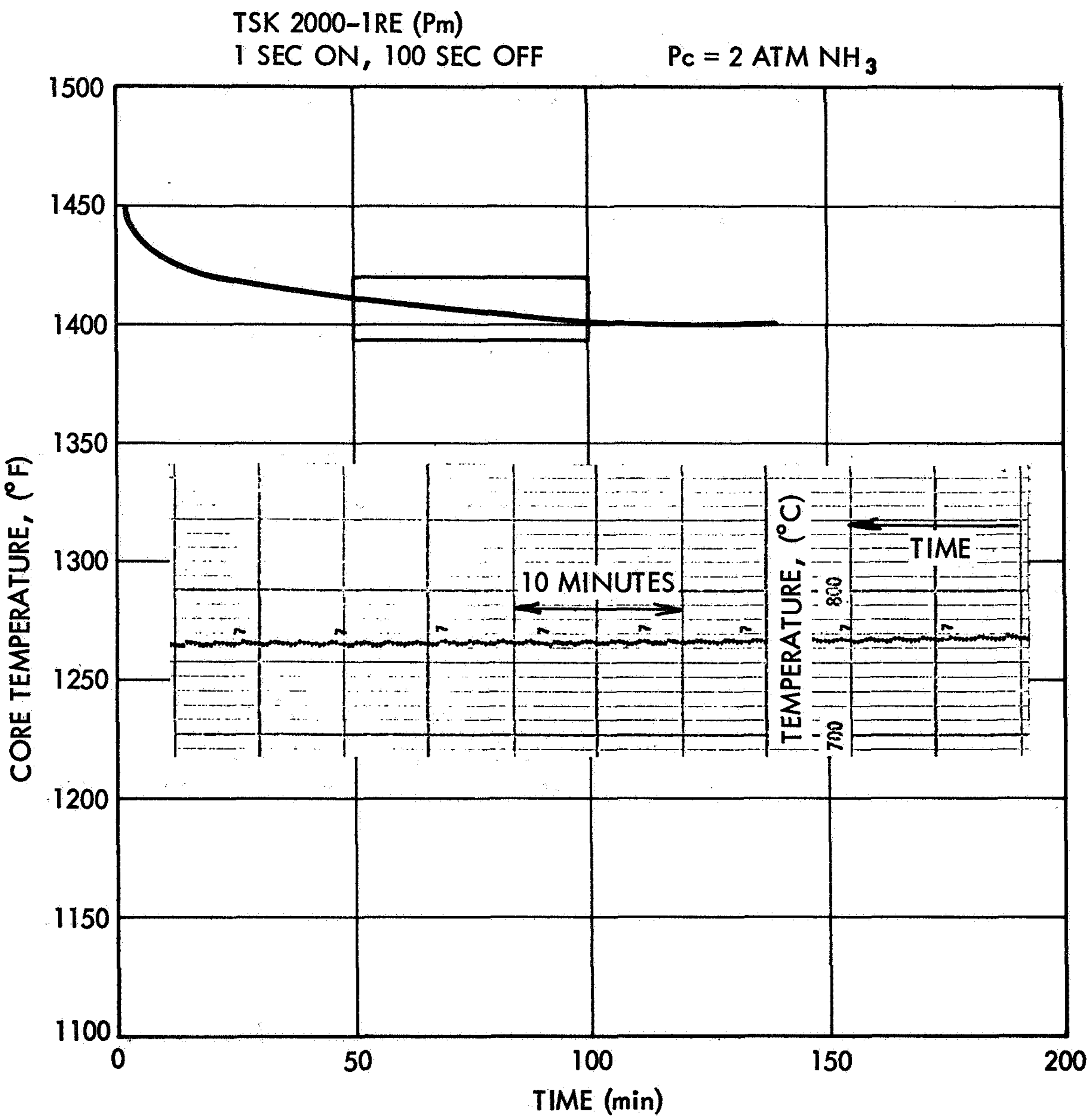
TSK 2000-1RE (Pm)
240 SEC ON
P_c = 1.5 ATM NH_3
CORE TEMPERATURE, (°F)
TIME (min)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
TEMPERATURE, (°C)
10 MINUTES
TIME
600
700
800

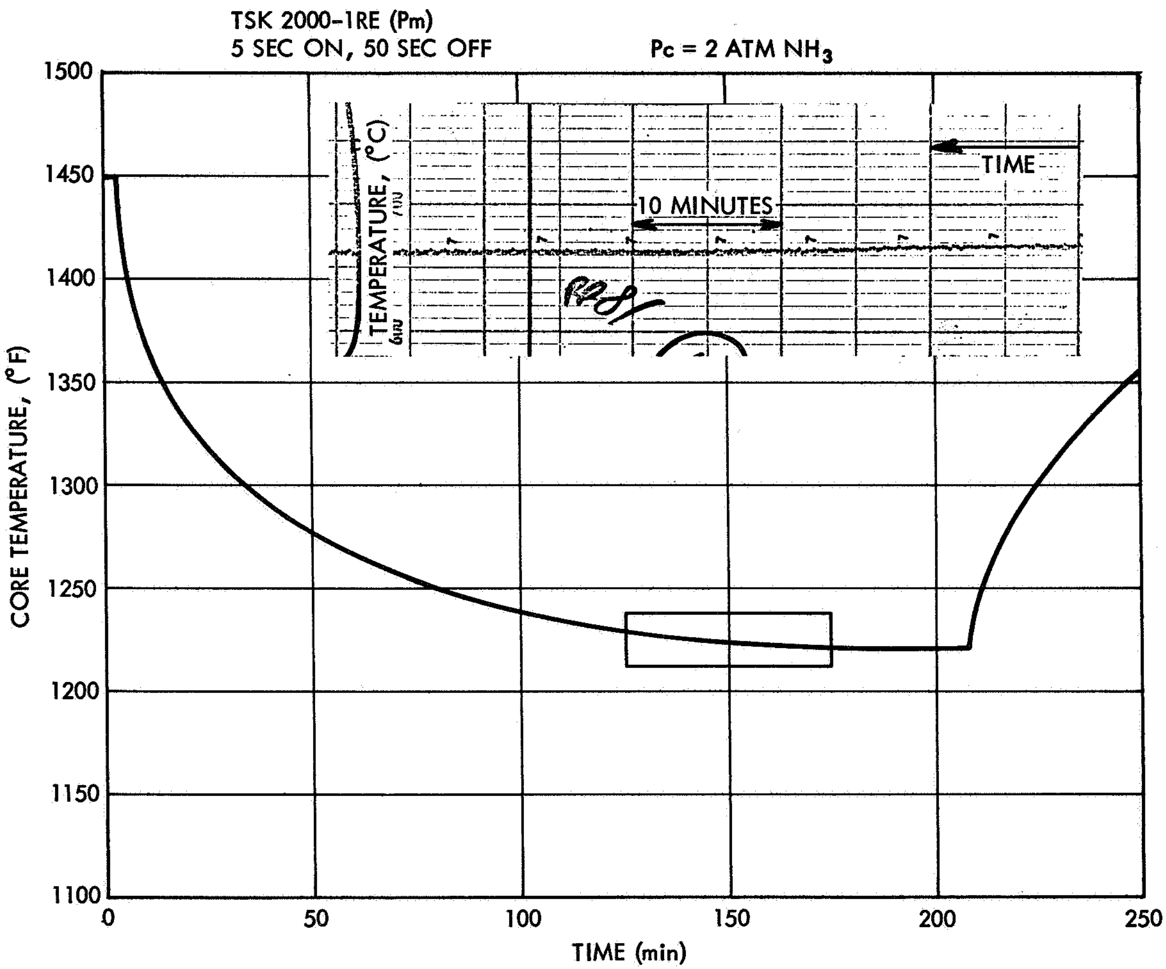

TSK 2000-1RE (Pm)
1 SEC ON, 1 SEC OFF
Pc = 2 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
800
700
600

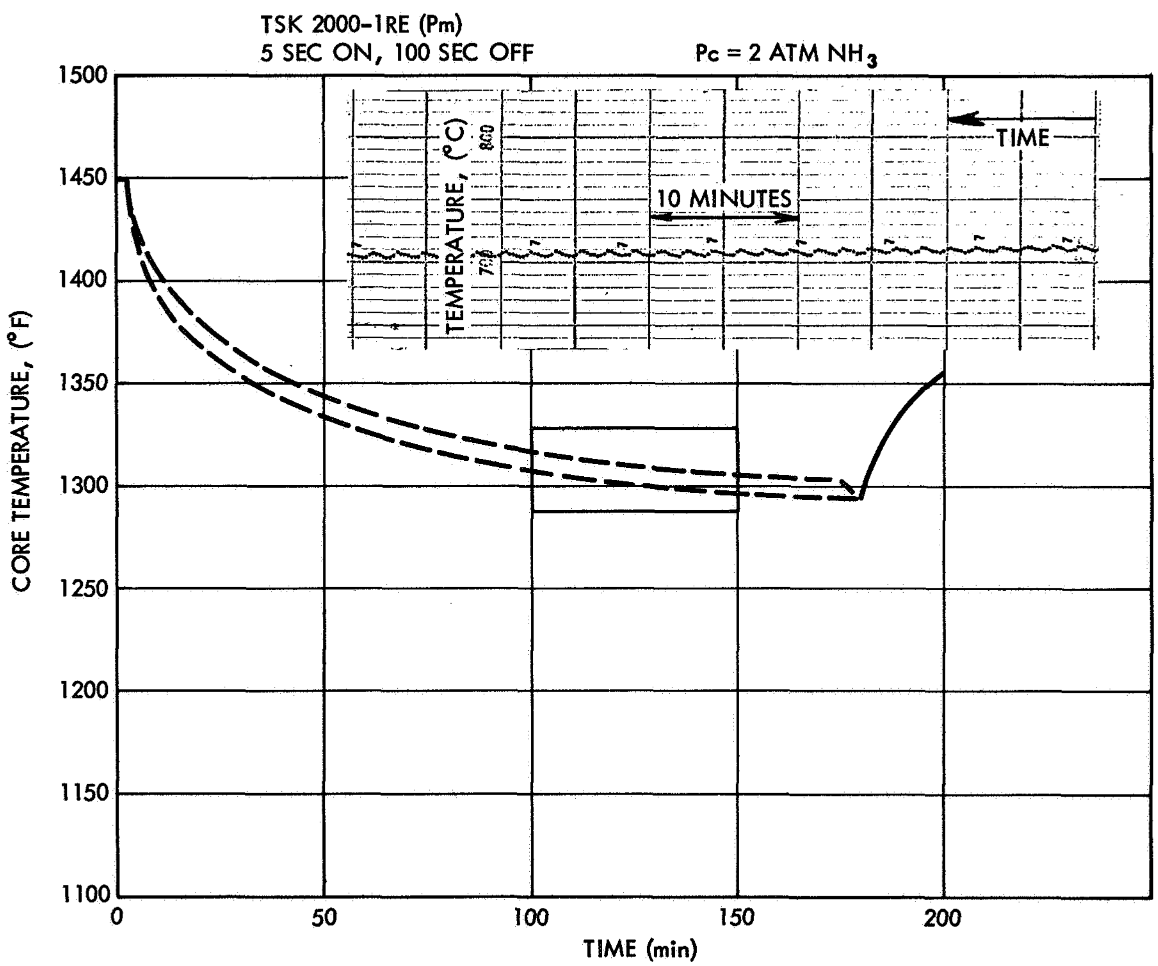

TSK 2000-1RE (Pm)
5 SEC ON, 5 SEC OFF
Pc = 2 ATM NH_3
1500
1400
1300
1200
1100
1000
900
800
CORE TEMPERATURE, (°F)
0
100
200
300
400
TIME (min)



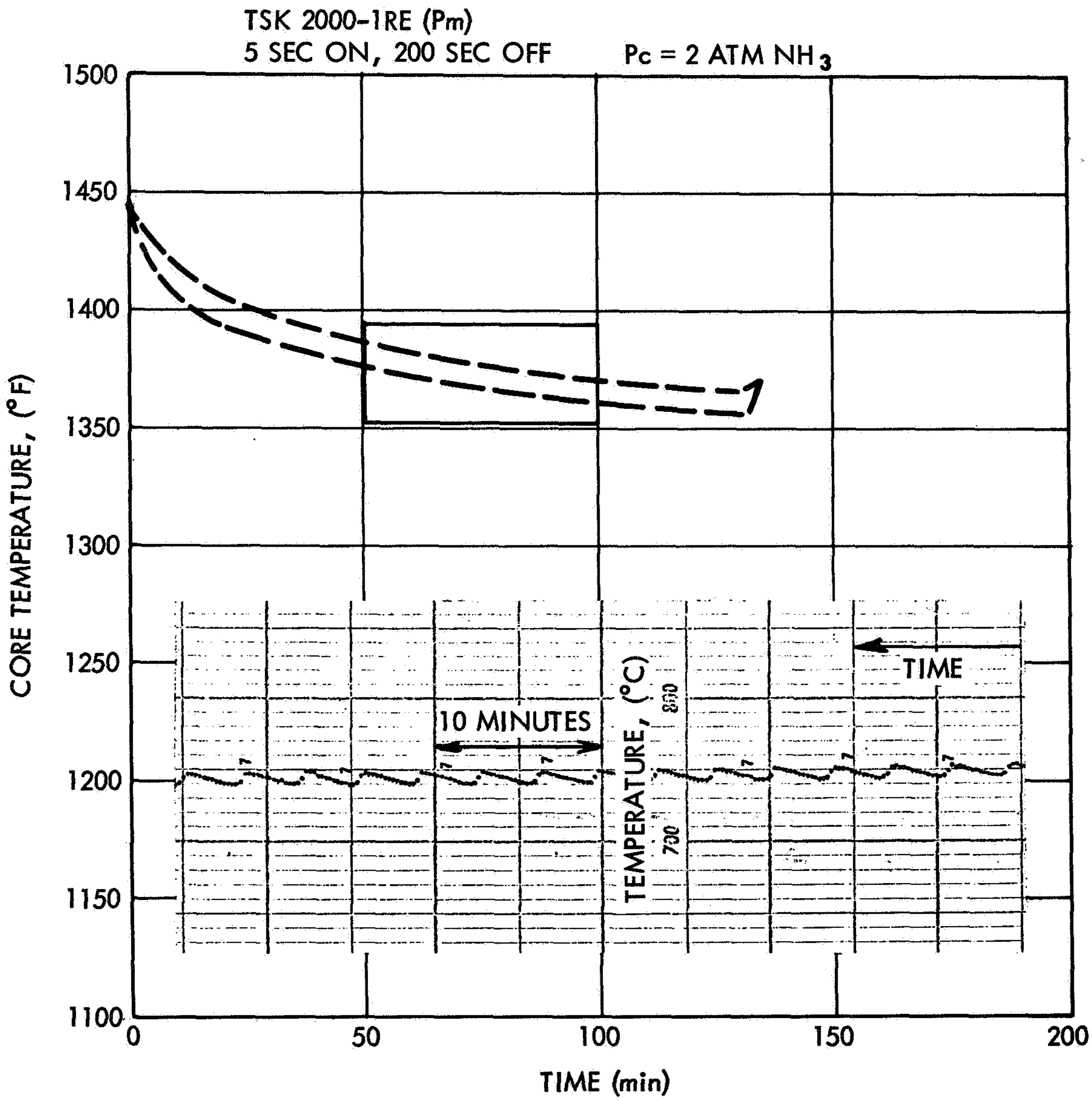
TSK 2000-1RE (Pm)
1 SEC ON, 50 SEC OFF
Pc = 2 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
800
700

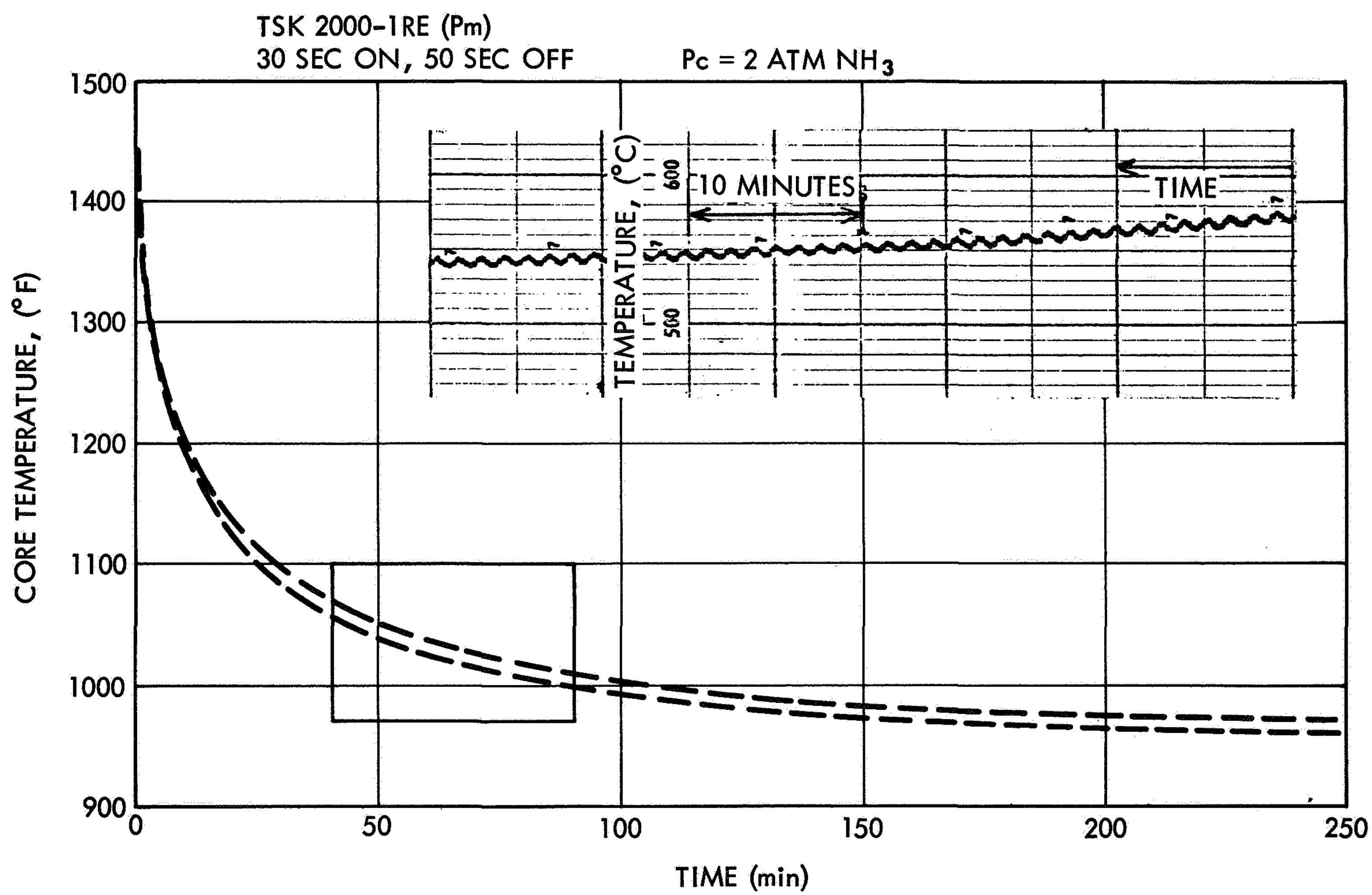

TSK 2000-1RE (Pm)
1 SEC ON, 100 SEC OFF
Pc = 2 ATM NH_3
1500
1450
1400
1350
1300
1250
1200
1150
1100
CORE TEMPERATURE, (°F)
0
50
100
150
200
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
800
700


TSK 2000-1RE (Pm)
5 SEC ON, 50 SEC OFF
Pc = 2 ATM NH3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
250
TIME (min)
TEMPERATURE, (°C)
TIME
10 MINUTES

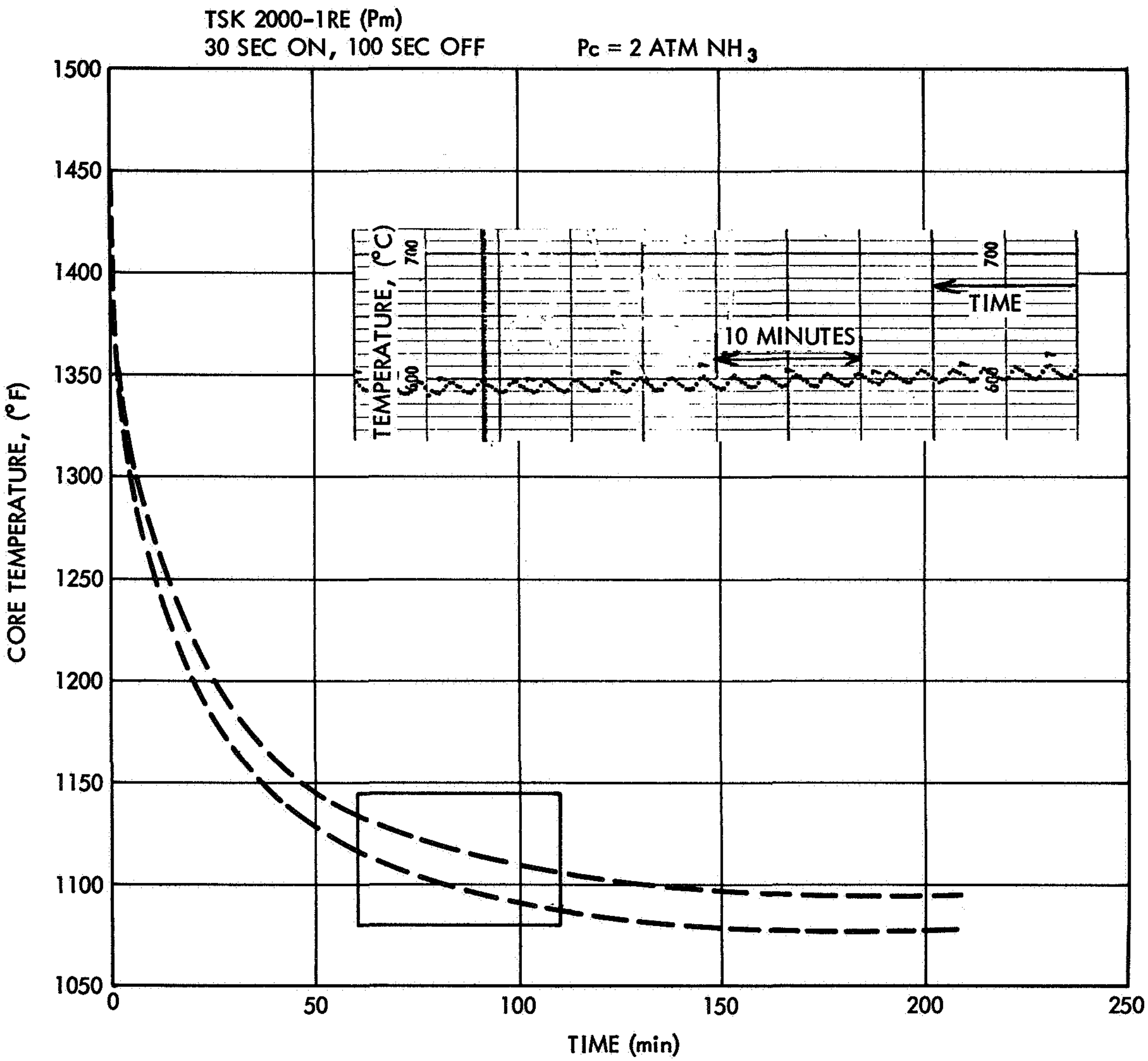


TSK 2000-1RE (Pm)
5 SEC ON, 100 SEC OFF
Pc = 2 ATM NH3
TIME
10 MINUTES
TEMPERATURE, (°C)
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
TIME (min)

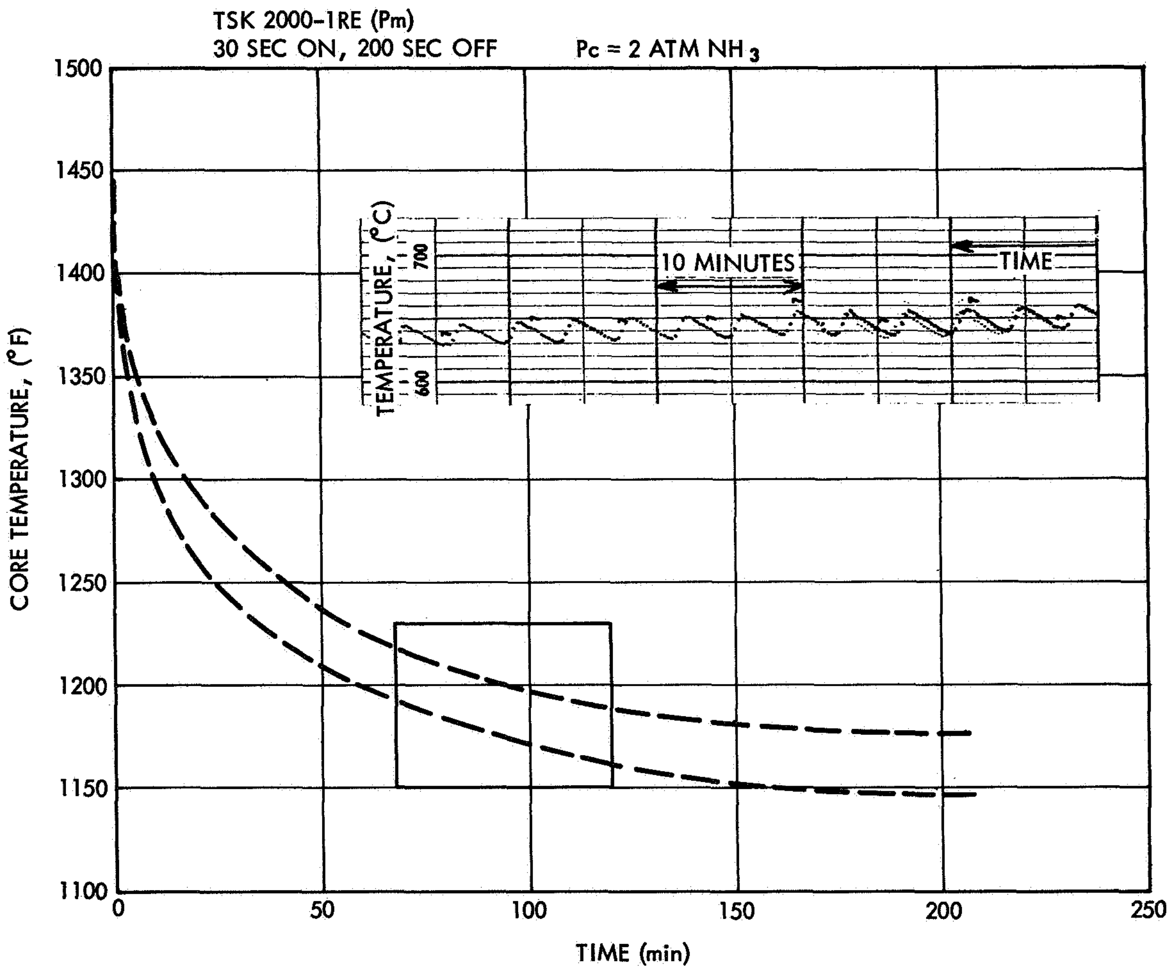

TSK 2000-1RE (Pm)
5 SEC ON, 200 SEC OFF
Pc = 2 ATM NH_3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
800
700



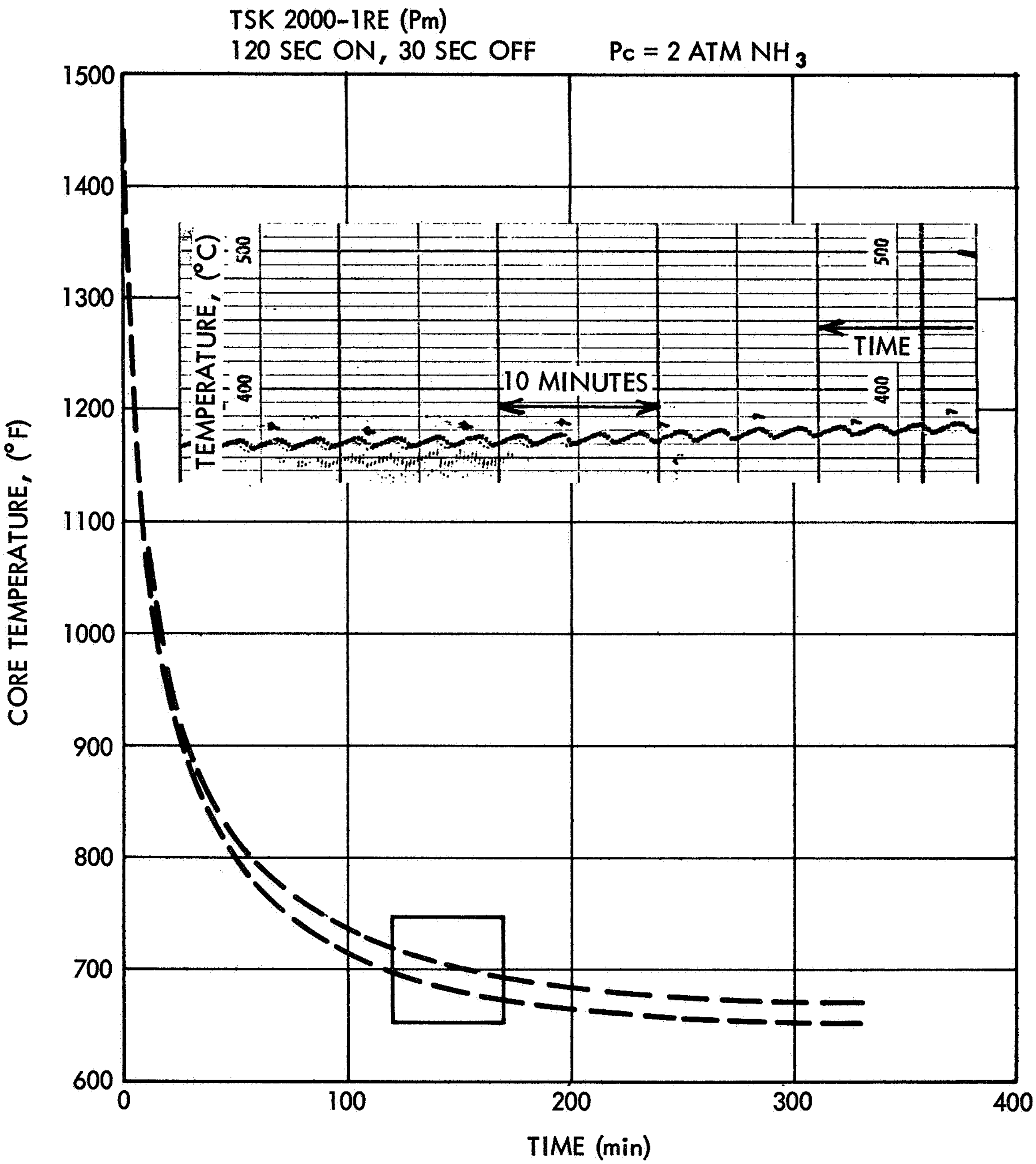
TSK 2000-1RE (Pm)
30 SEC ON, 50 SEC OFF
Pc = 2 ATM NH3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
0
50
100
150
200
250
TIME (min)
TEMPERATURE, (°C)
600
500
10 MINUTES
TIME



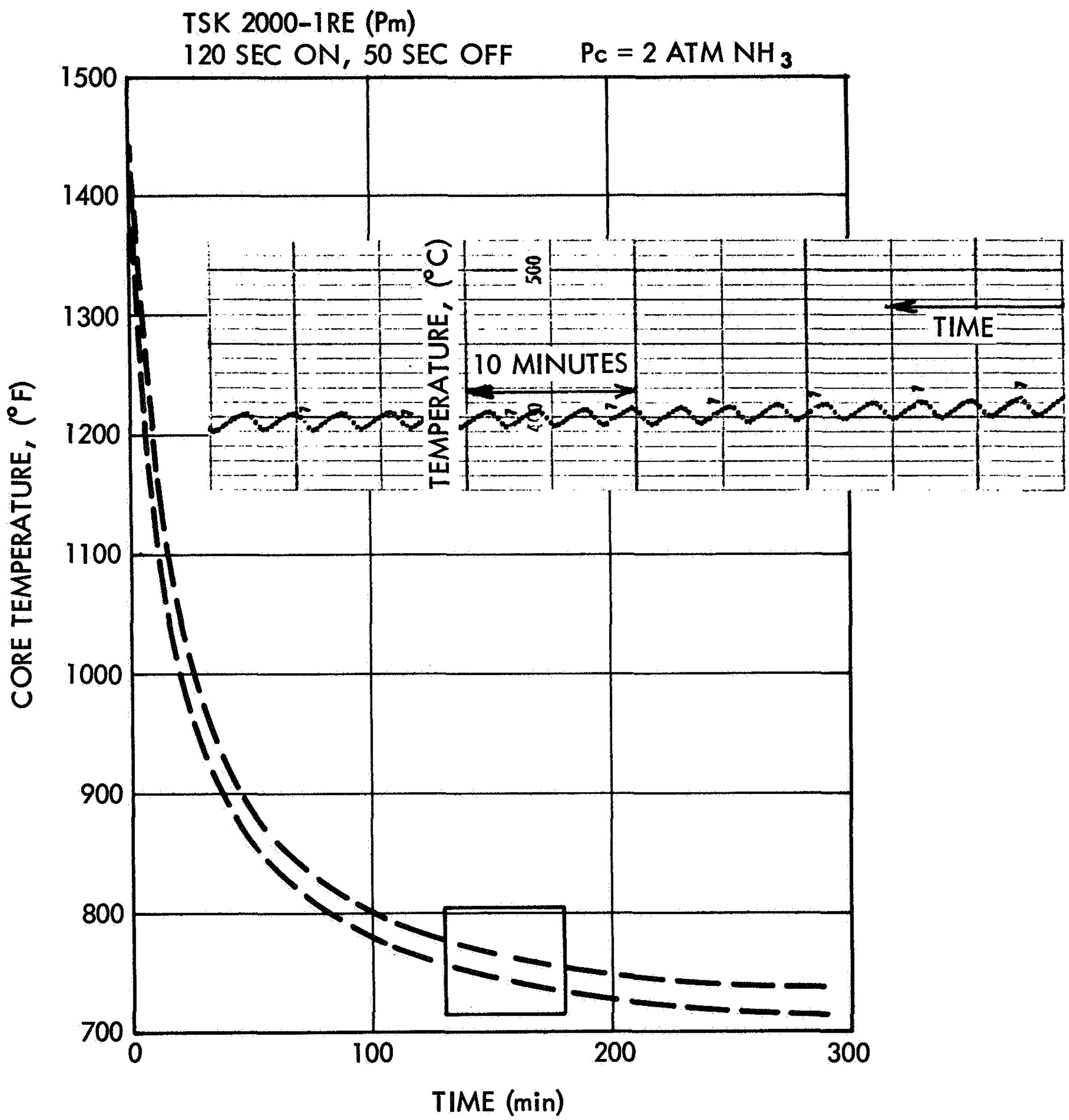
TSK 2000-1RE (Pm)
30 SEC ON, 100 SEC OFF
Pc = 2 ATM NH_3
CORE TEMPERATURE, (°F)
TIME (min)
1500
1450
1400
1350
1300
1250
1200
1150
1100
1050
0
50
100
150
200
250
TEMPERATURE, (°C)
700
600
TIME
10 MINUTES



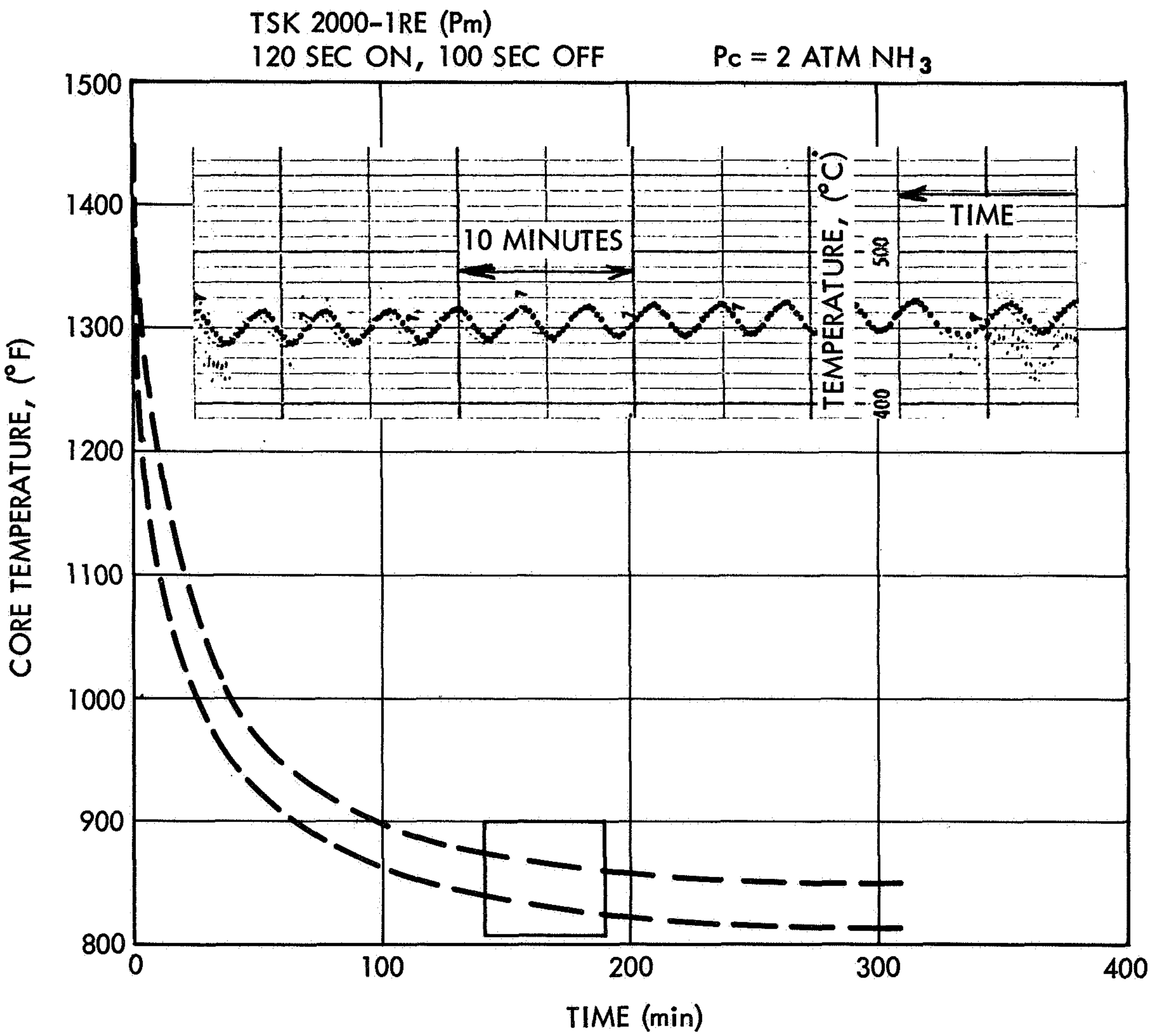
TSK 2000-1RE (Pm)
30 SEC ON, 200 SEC OFF
Pc = 2 ATM NH 3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
50
100
150
200
250
TIME (min)
TEMPERATURE, (°C)
700
600
10 MINUTES
TIME

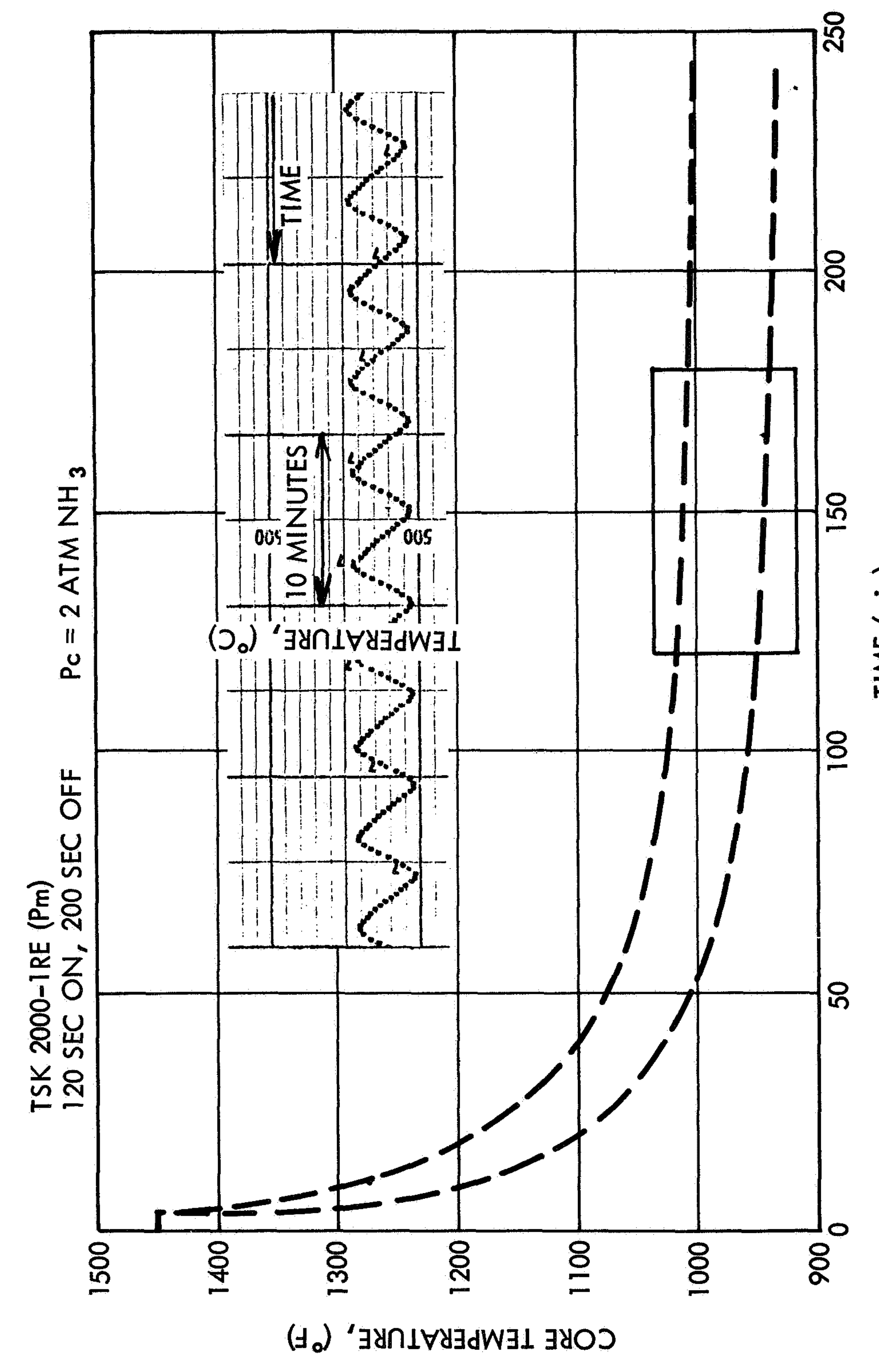


TSK 2000-1RE (Pm)
120 SEC ON, 30 SEC OFF
Pc = 2 ATM NH3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
800
700
600
TIME (min)
0
100
200
300
400
TEMPERATURE, (°C)
500
400
10 MINUTES
TIME

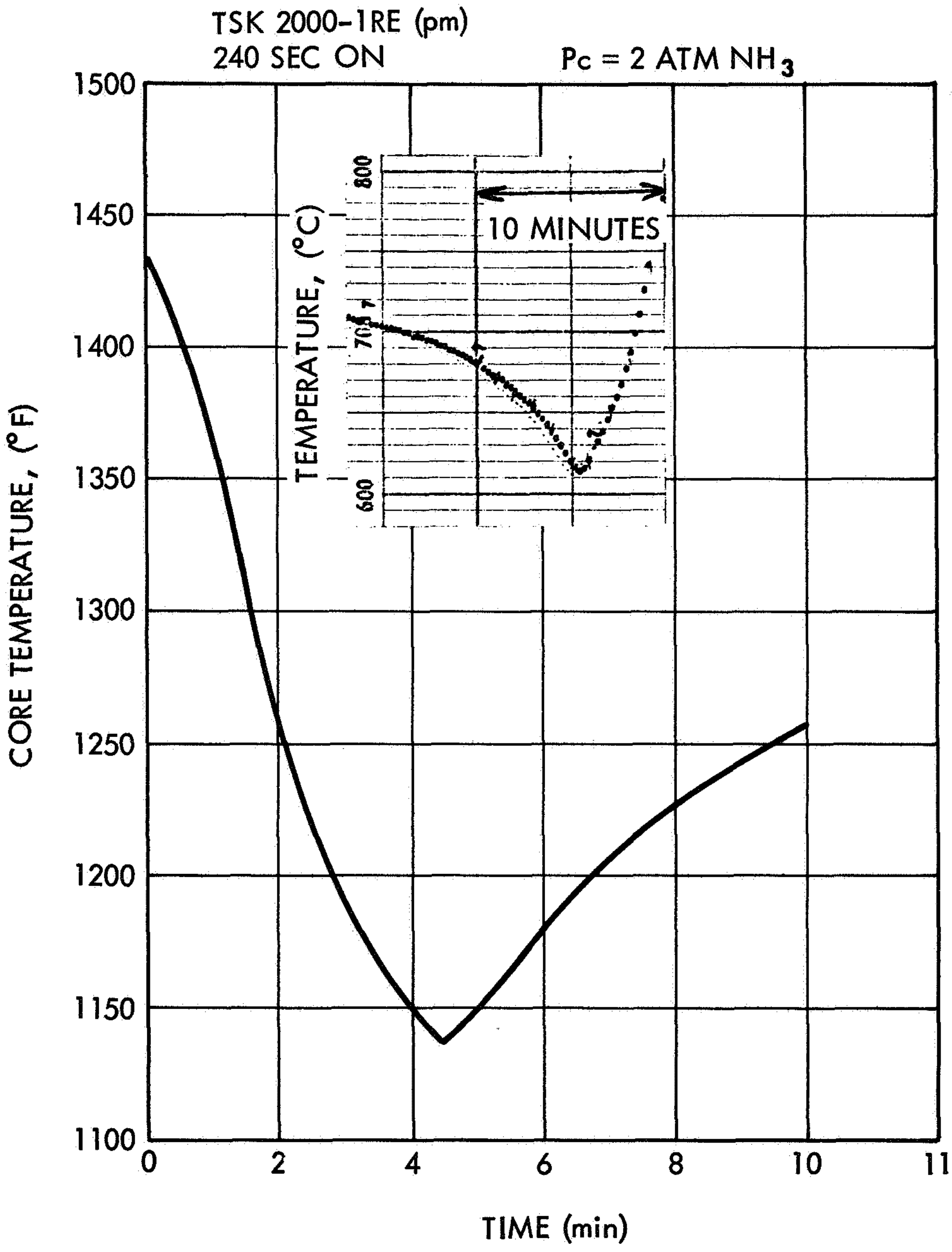


TSK 2000-1RE (Pm)
120 SEC ON, 50 SEC OFF
Pc = 2 ATM NH_3
1500
1400
1300
1200
1100
1000
900
800
700
CORE TEMPERATURE, (°F)
0
100
200
300
TIME (min)
TEMPERATURE, (°C)
500
400
TIME
10 MINUTES

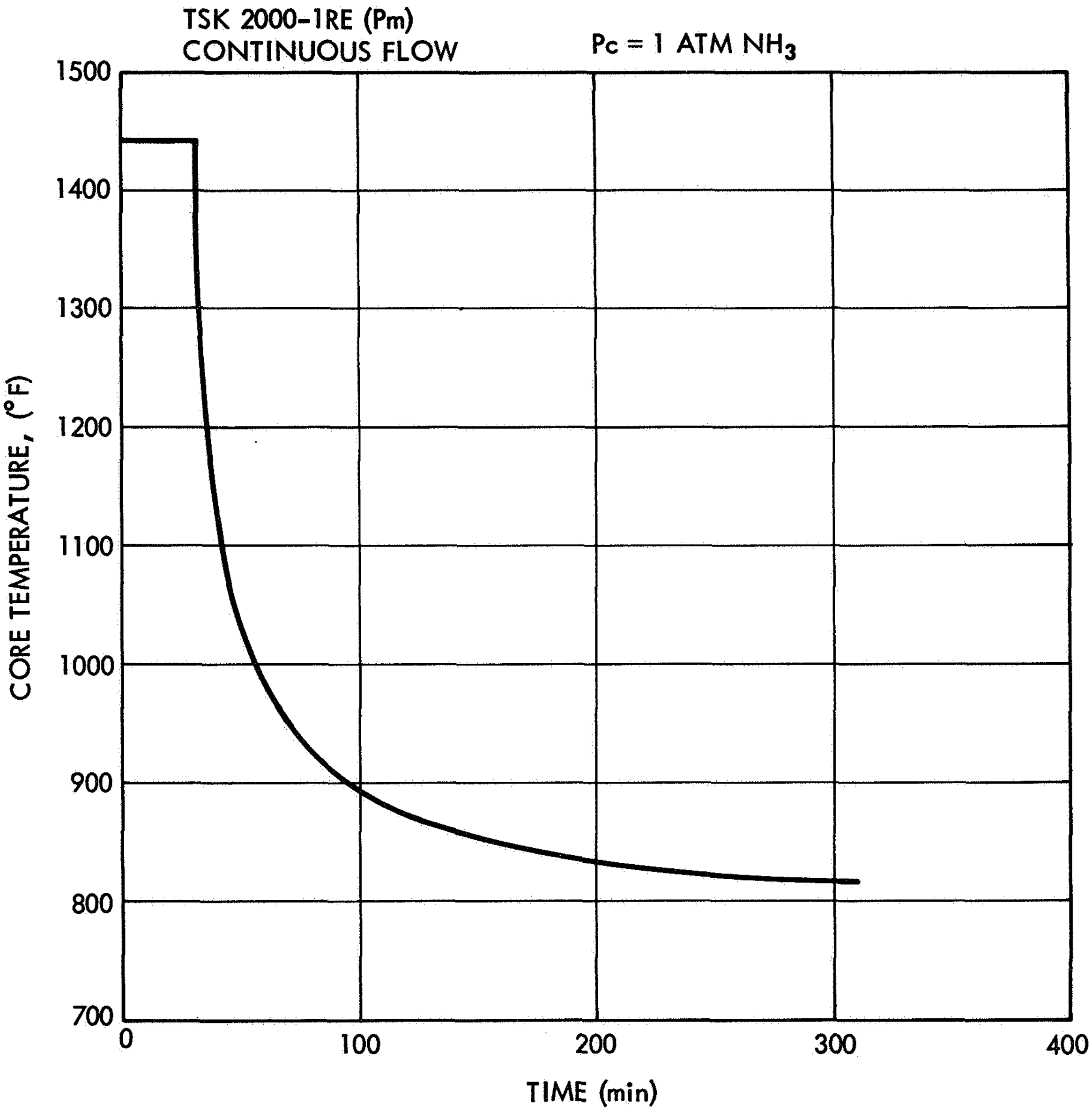

TSK 2000-1RE (Pm)
120 SEC ON, 100 SEC OFF
Pc = 2 ATM NH3
1500
1400
1300
1200
1100
1000
900
800
CORE TEMPERATURE, (°F)
0
100
200
300
400
TIME (min)
10 MINUTES
TIME
TEMPERATURE, (°C)
500
400

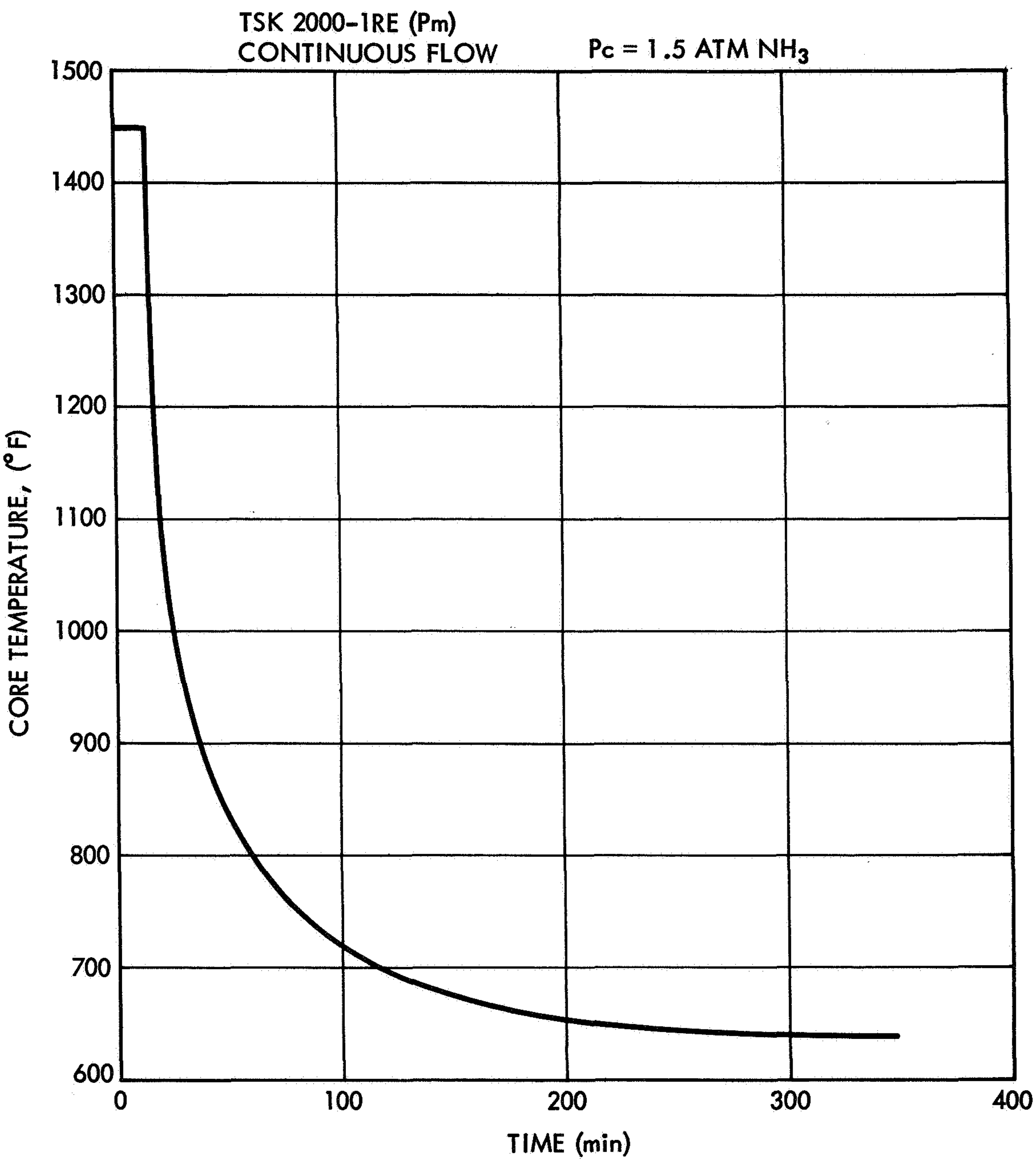


TSK 2000-1RE (Pm)
120 SEC ON, 200 SEC OFF
Pc = 2 ATM NH_3
TEMPERATURE, (°C)
10 MINUTES
TIME
500
CORE TEMPERATURE, (°F)
TIME (min)
1500
1400
1300
1200
1100
1000
900
0
50
100
150
200
250

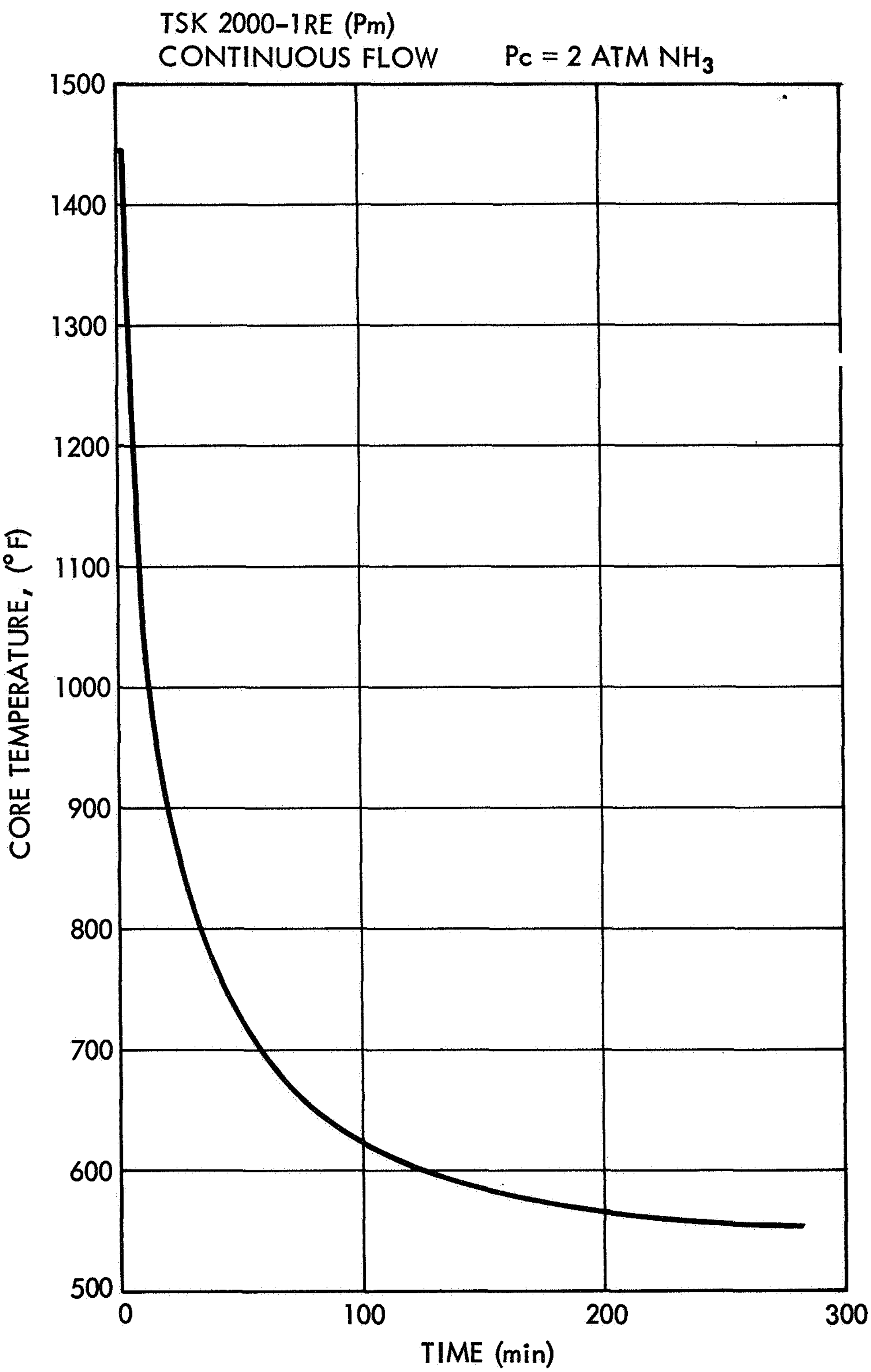


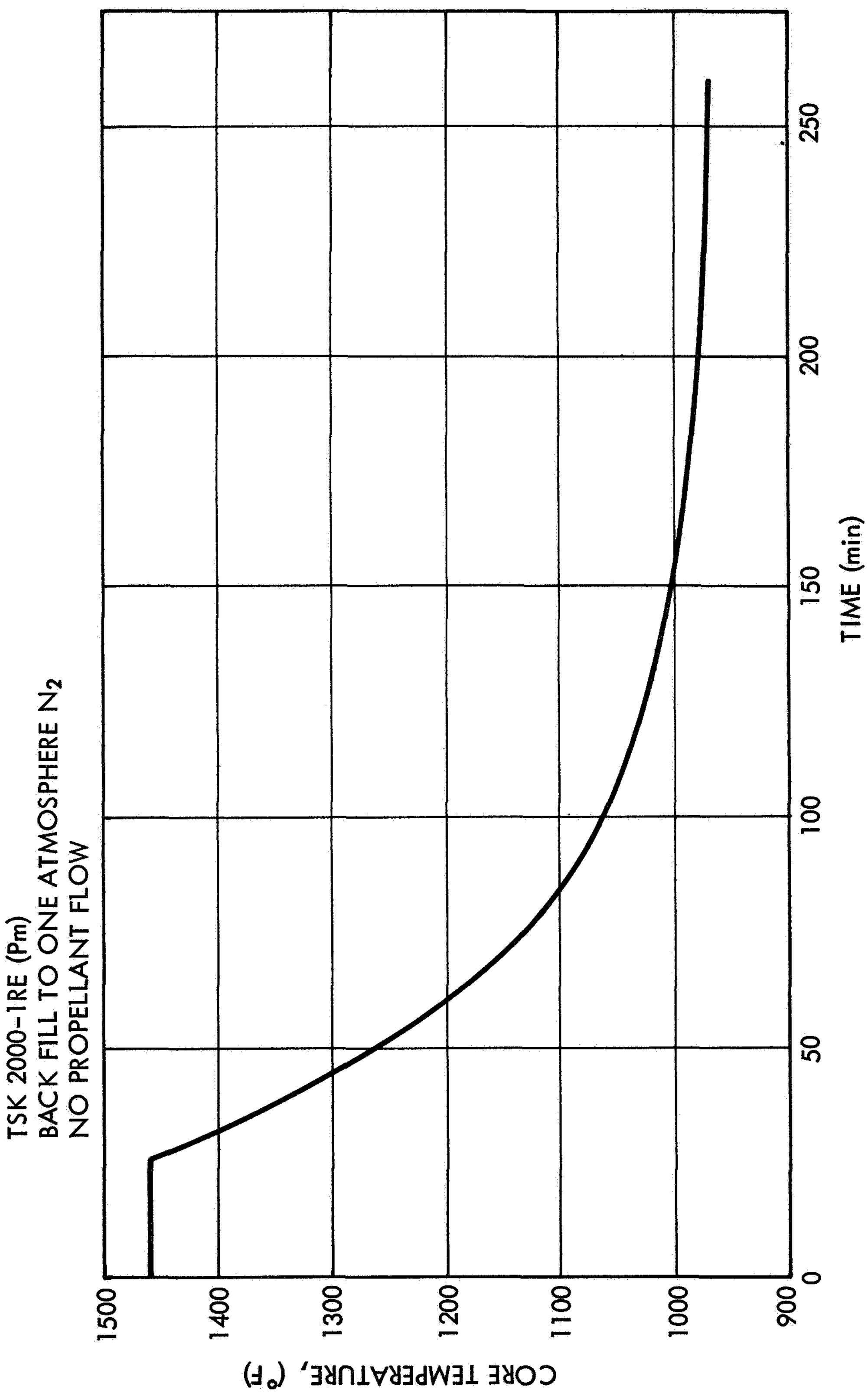
TSK 2000-1RE (pm)
240 SEC ON
Pc = 2 ATM NH_3
CORE TEMPERATURE, (°F)
1500
1450
1400
1350
1300
1250
1200
1150
1100
0
2
4
6
8
10
11
TIME (min)
TEMPERATURE, (°C)
800
700
600
10 MINUTES

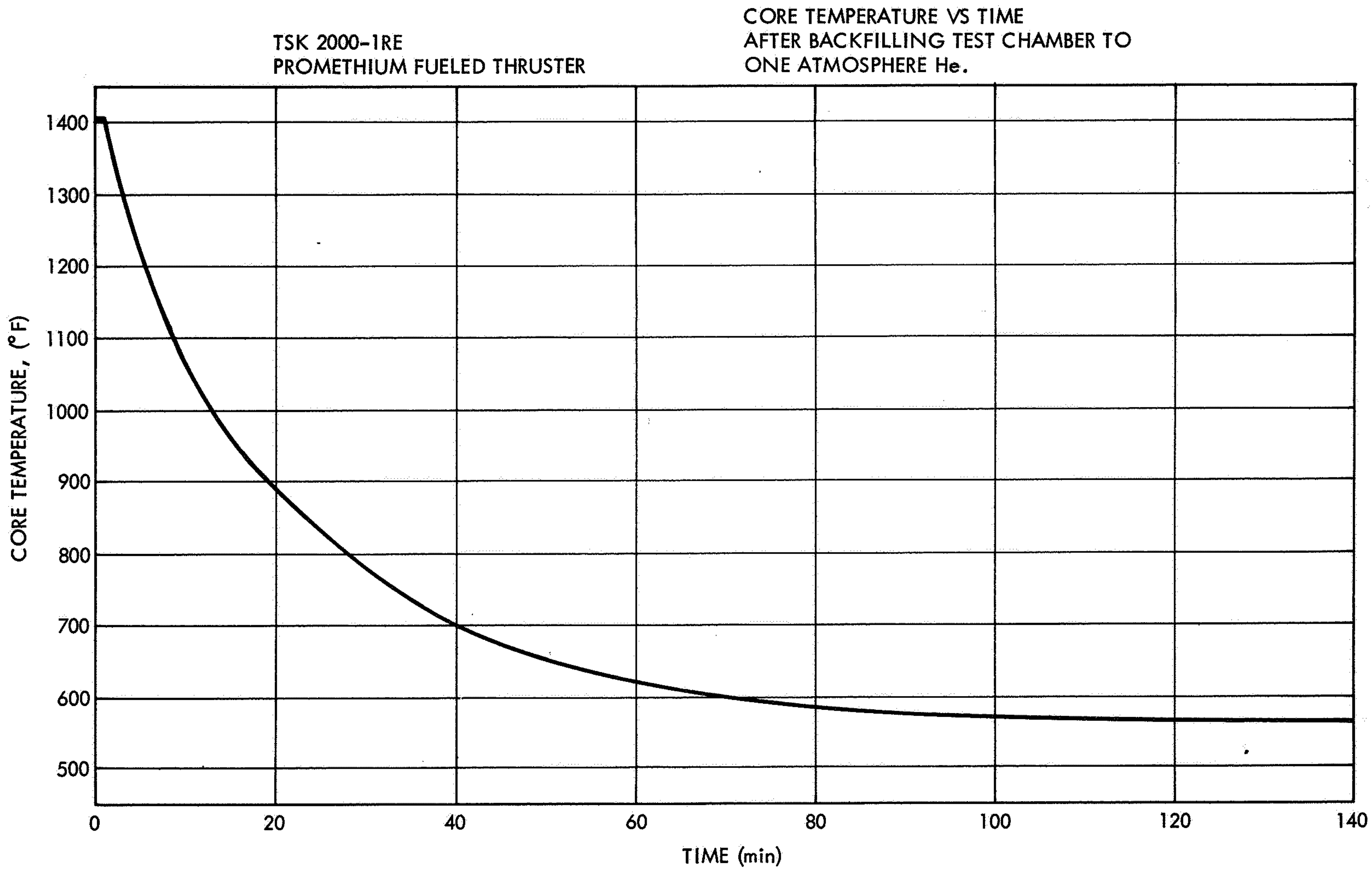

TSK 2000-1RE (Pm)
CONTINUOUS FLOW
Pc = 1 ATM NH3
1500
1400
1300
1200
1100
1000
900
800
700
CORE TEMPERATURE, (°F)
0
100
200
300
400
TIME (min)



TSK 2000-1RE (Pm)
CONTINUOUS FLOW
Pc = 1.5 ATM NH_3
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
800
700
600
0
100
200
300
400
TIME (min)


TSK 2000-1RE (Pm)
CONTINUOUS FLOW
Pc = 2 ATM NH_3
1500
1400
1300
1200
1100
1000
900
800
700
600
500
CORE TEMPERATURE, (°F)
0
100
200
300
TIME (min)


TSK 2000-1RE (Pm)
BACK FILL TO ONE ATMOSPHERE N2
NO PROPELLANT FLOW
CORE TEMPERATURE, (°F)
1500
1400
1300
1200
1100
1000
900
TIME (min)
0
50
100
150
200
250


TSK 2000-1RE
PROMETHIUM FUELED THRUSTER
CORE TEMPERATURE VS TIME
AFTER BACKFILLING TEST CHAMBER TO
ONE ATMOSPHERE He.
CORE TEMPERATURE, (°F)
1400
1300
1200
1100
1000
900
800
700
600
500
0
20
40
60
80
100
120
140
TIME (min)